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WL-TR-97-4073

PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES REVIEW (9^{TH})

Sponsored by:

Air Force Wright Aeronautical Laboratories
Materials Laboratory

APRIL 1997

FINAL REPORT FOR PERIOD 24-26 OCTOBER 1983

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MATERIALS DIRECTORATE
WRIGHT LABORATORY
AIR FORCE MATERIEL COMMAND
WRIGHT-PATTERSON AFB OH 45433-7734

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13. ABSTRACT *(Maximum 200 words)*

This report contains the basic unedited vu-graphs of the presentations at the "Mechanics" of Composites Review" sponsored jointly by the Non-metallic Materials Division of the Air Force Materials Laboratory, the Structures Division of the Air Force Flight Dynamics Laboratory and the Directorate of Aerospace Sciences of the Air Force Office of Scientific Research. The presentations cover current in-house and contract programs under the sponsorship of these three organizations.

14. SUBJECT TERMS

epoxy-matrix composites; composite materials; resin matrix composites; composite bonded joints; fatigue of graphite/epoxy composites; fracture and fatigue of bi-materials

15. NUMBER OF PAGES

277

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
UNCLASSIFIED	UNCLASSIFIED	UNCLASSIFIED	SAR

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AGENDA

MECHANICS OF COMPOSITES REVIEW

24-26 OCTOBER 1983

FOREWORD

This report contains the abstracts and viewgraphs of the presentations at the Ninth Annual Mechanics of Composites Review sponsored by the Materials Laboratory. Each was prepared by its presenter and is published here unedited. In addition, a listing of both the in-house and contractual activities of each participating organization is included.

The Mechanics of Composites Review is designed to present programs covering activities throughout the United States Air Force, Army, Navy, and NASA. Programs not covered in the present review are candidates for presentation at future mechanics of composites reviews. The presentations cover both in-house and contract programs under the sponsorship of the participating organizations.

Since this is a review of on-going programs, much of the information in this report has not been published as yet and is subject to change; but timely dissemination of the rapidly expanding technology of advanced composites is deemed highly desirable. Works in the area of mechanics of composites have long been typified by disciplined approaches. It is hoped that such a high standard of rigor is reflected in the majority, if not all, of the presentations in this report.

Feedback and open critique of the presentations and the review itself are most welcome as suggestions and recommendations from all participants will be considered in the planning of future reviews.

Thanks are due to Lisa Wilson for her planning and implementation of the Review.

F. D. Cherry

FRANKLIN D. CHERRY, Chief
Nonmetallic Materials Division
Materials Laboratory

ACKNOWLEDGEMENT

We express our appreciation to the authors for their contributions and to the points of contact within the organizations for their efforts in supplying the program listings.

DURABILITY OF COMPOSITES

R.S. WHITEHEAD/G.L. RITCHIE/J.L. MULLINEAUX

CONTRACT: WING FUSELAGE CRITICAL COMPONENT DEVELOPMENT PROGRAM
NUMBER: F33615-79-C-3203
SPONSOR: AFWAL/FIBAC

AIR FORCE PROJECT ENGINEER: J.L. MULLINEAUX
NORTHROP PROGRAM MANAGER: G.L. RITCHIE

PROGRAM OBJECTIVES

DEVELOP GENERIC CERTIFICATION PROCEDURES TO PROVIDE HIGH CONFIDENCE LEVEL IN COMPOSITE PRIMARY STRUCTURE

- DESIGN TECHNOLOGY
- DURABILITY METHODOLOGY
- ADVANCED MANUFACTURING/ PRODUCTION COST

79-13143A
10

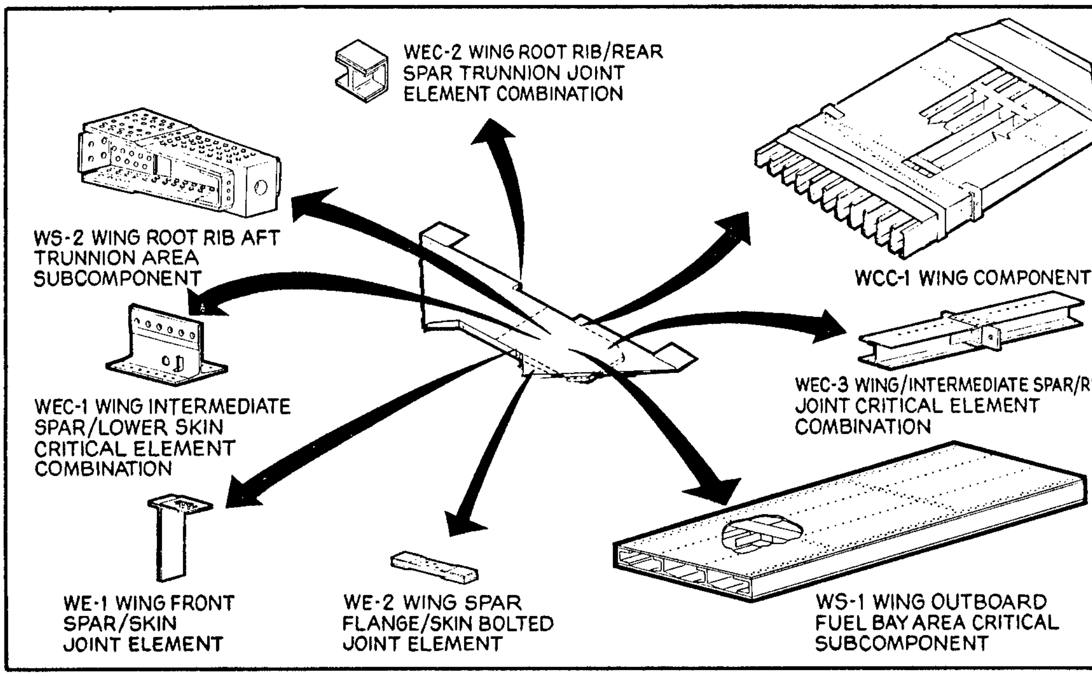
CONCLUSIONS

- RT/DRY FATIGUE DOES NOT DEGRADE IN-PLANE LOADED COMPOSITE LAMINATES
- STATIC STRENGTH OF 3501-6 LAMINATES IN COMPRESSION AT 250 F/WET IS REDUCED UP TO 35%
- SEVERE ENVIRONMENTAL FATIGUE REDUCES STRENGTH OF 3501-6 LAMINATES UP TO 17 %
- OUT-OF-PLANE LOADING CAUSES GREATEST STRENGTH REDUCTIONS
- LOW-COST CERTIFICATION PROCEDURES ARE PRACTICAL IF ENVIRONMENTAL CONDITIONS ARE NOT SEVERE

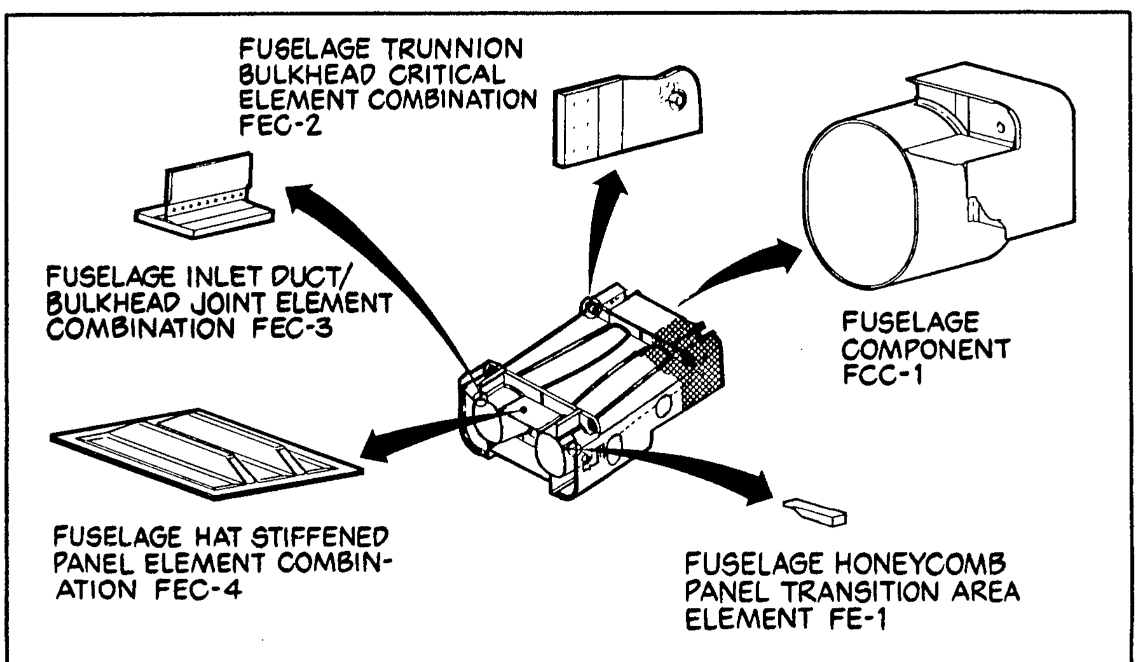
DESIGN AND FABRICATION APPROACH

- SELECT MOST CRITICAL LOCATIONS IN HIGH PERFORMANCE AIRCRAFT WING/FUSELAGE PRIMARY STRUCTURE
- DESIGN TEST SPECIMENS REPRESENTATIVE OF CRITICAL LOCATIONS
  - 12 COUPONS, 4 ELEMENTS, 7 ELEMENT COMBINATIONS
  - 2 SUBCOMPONENTS
  - 2 FULL-SCALE COMPONENTS
- FABRICATE TEST SPECIMENS
  - DESIGN AND FABRICATE TOOLS
  - FAB IN PRODUCTION SHOP WITH FULL QUALITY ASSURANCE

WING TEST SPECIMENS



FUSELAGE TEST SPECIMENS



DESIGN AND FABRICATION APPROACH

- DEVELOP DURABILITY AND DAMAGE TOLERANCE TEST PROCEDURES
  - RT AND HOT/WET STATIC
  - RT AND ENVIRONMENTAL BASELINE ACCELERATED FATIGUE
  - ALTERNATE ENVIRONMENTAL ACCELERATED FATIGUE
  - ENVIRONMENTAL REAL-TIME FATIGUE
- PERFORM TESTS
- ANALYZE TEST DATA
  - DEVELOP DURABILITY DESIGN METHODOLOGY
  - DEVELOP DURABILITY CERTIFICATION PROCEDURES

TEST MATRIX

<table>
<tr><th rowspan="3">SPECIMEN</th><th colspan="7">TEST SERIES</th></tr>
<tr><th colspan="2">STATIC</th><th colspan="4">ACCELERATED FATIGUE</th><th rowspan="2">REAL FLIGHT TIME</th></tr>
<tr><th>250 F/WET</th><th>AMBIENT</th><th>AMBIENT</th><th>BASELINE</th><th>ALTERNATE 1</th><th>ALTERNATE 2</th></tr>
<tr><td>COUPONS</td><td>55</td><td>55</td><td>55</td><td>55</td><td>12</td><td>12</td><td>0</td></tr>
<tr><td>ELEMENTS</td><td>9</td><td>9</td><td>9</td><td>9</td><td>3</td><td>3</td><td>0</td></tr>
<tr><td>ELEMENT COMBINATIONS</td><td>18</td><td>18</td><td>18</td><td>18</td><td colspan="2">6</td><td>4</td></tr>
<tr><td>SUBCOMPONENTS</td><td>5</td><td>4</td><td>6</td><td>5</td><td colspan="2">3</td><td>2</td></tr>
<tr><td>COMPONENTS</td><td>2</td><td>2</td><td>2</td><td>2</td><td colspan="2">2</td><td>2</td></tr>
<tr><td>TOTAL</td><td>89</td><td>88</td><td>90</td><td>89</td><td colspan="2">64</td><td>8</td></tr>
</table>

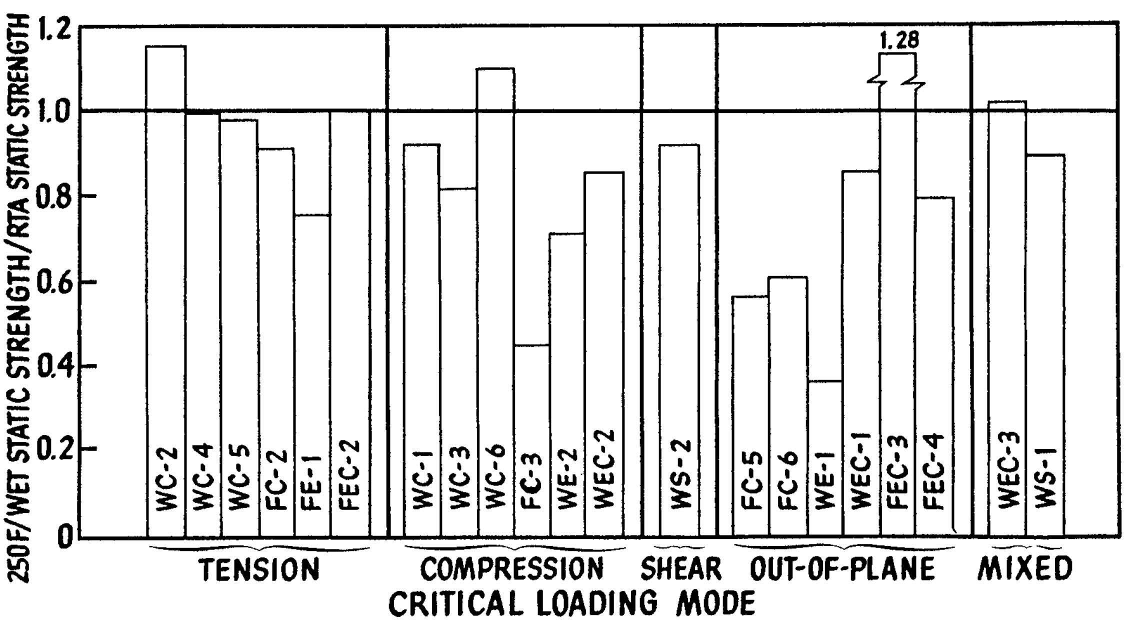
GRAND TOTAL = 428

83-11003
10

STATIC TESTS

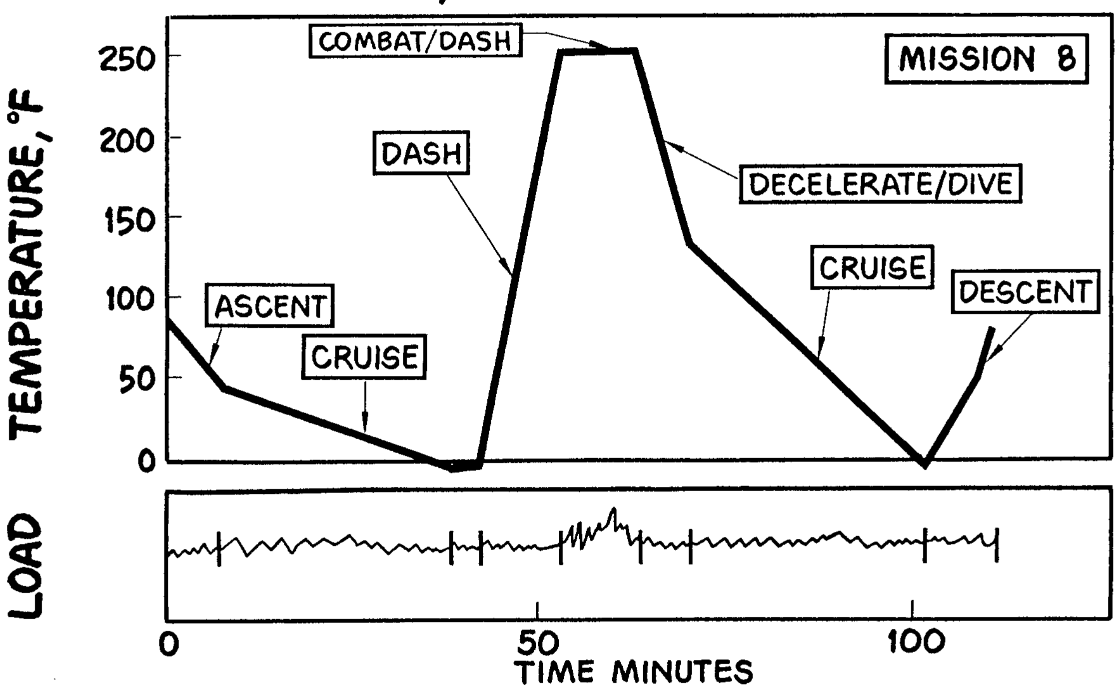
- ROOM TEMPERATURE AMBIENT
- 250F/WET
  (WET= END OF LIFETIME MOISTURE CONTENT)

250F/WET STATIC DATA

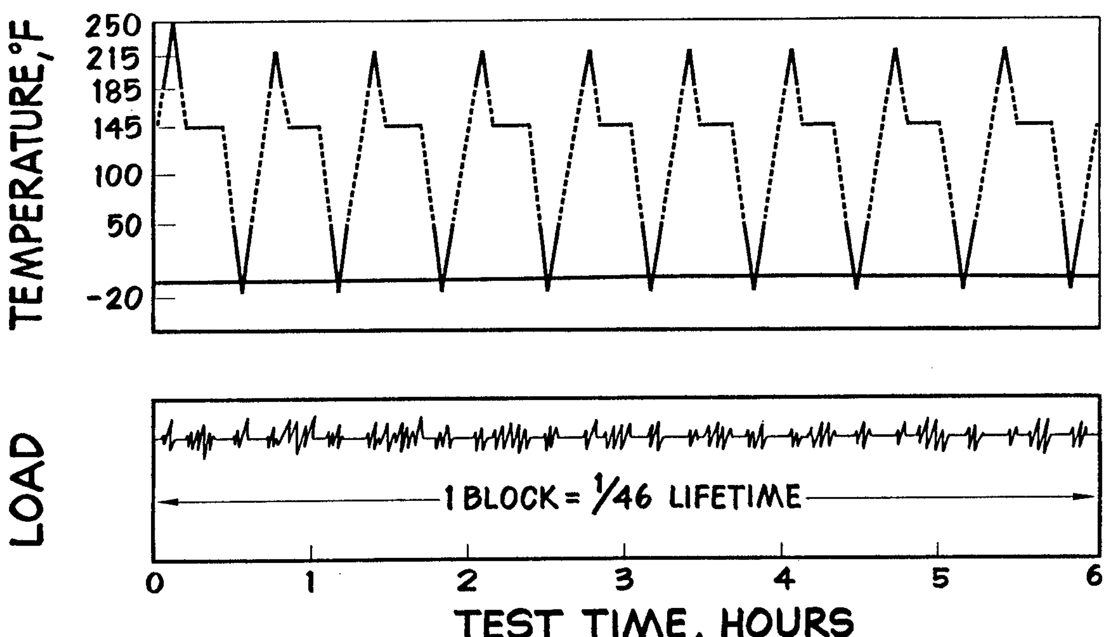


83-11006
10

REAL TIME LOAD/TEMPERATURE PROFILE



BASELINE ACCELERATED SCHEME



81-12774A
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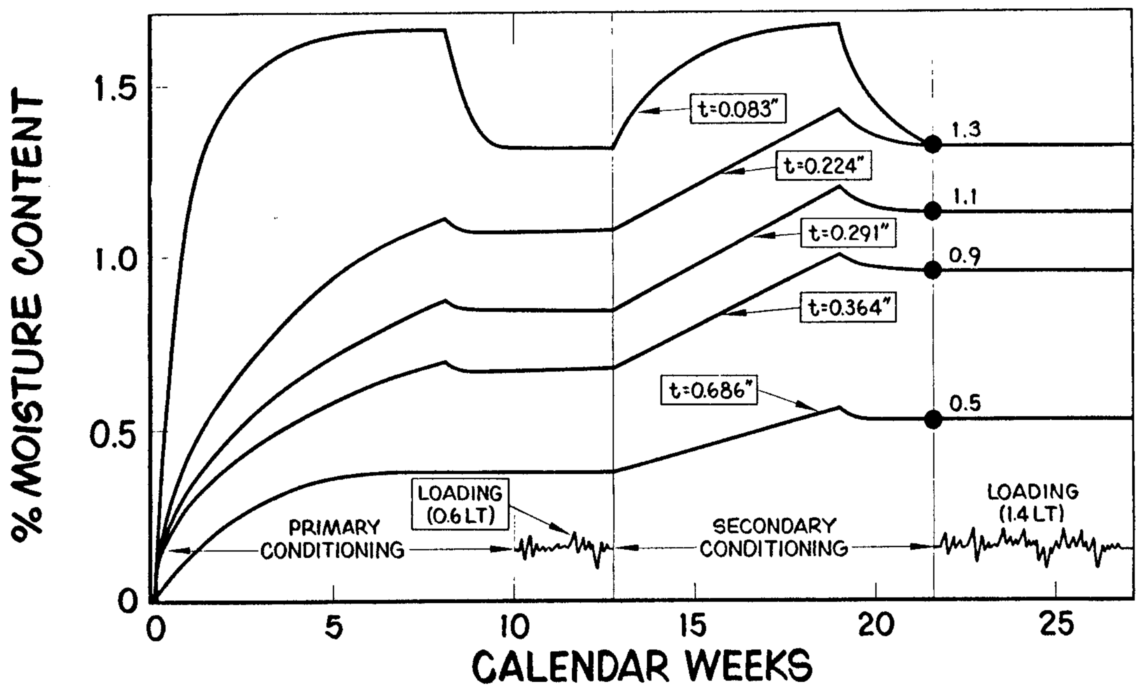
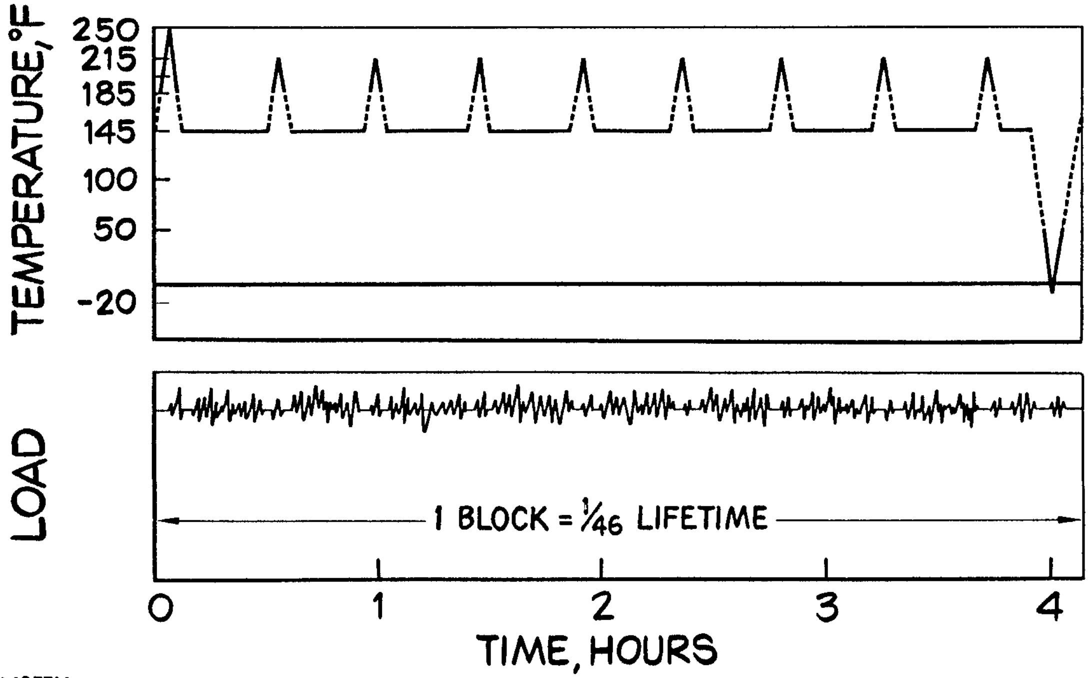
BASELINE ACCELERATED SCHEME

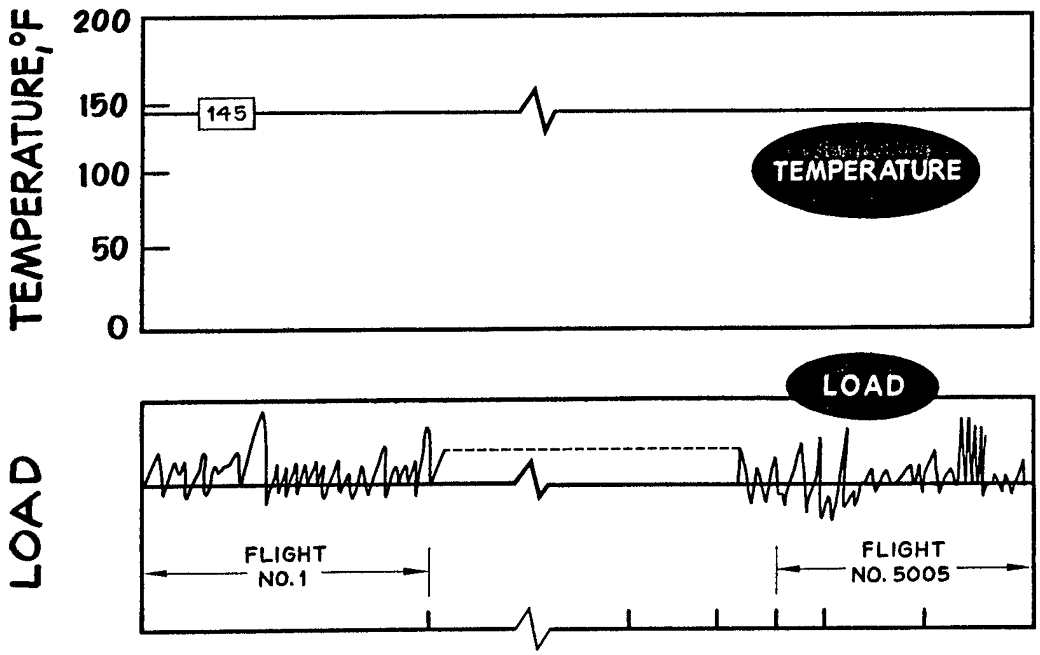


ALTERNATE TEST SCHEME No. 1



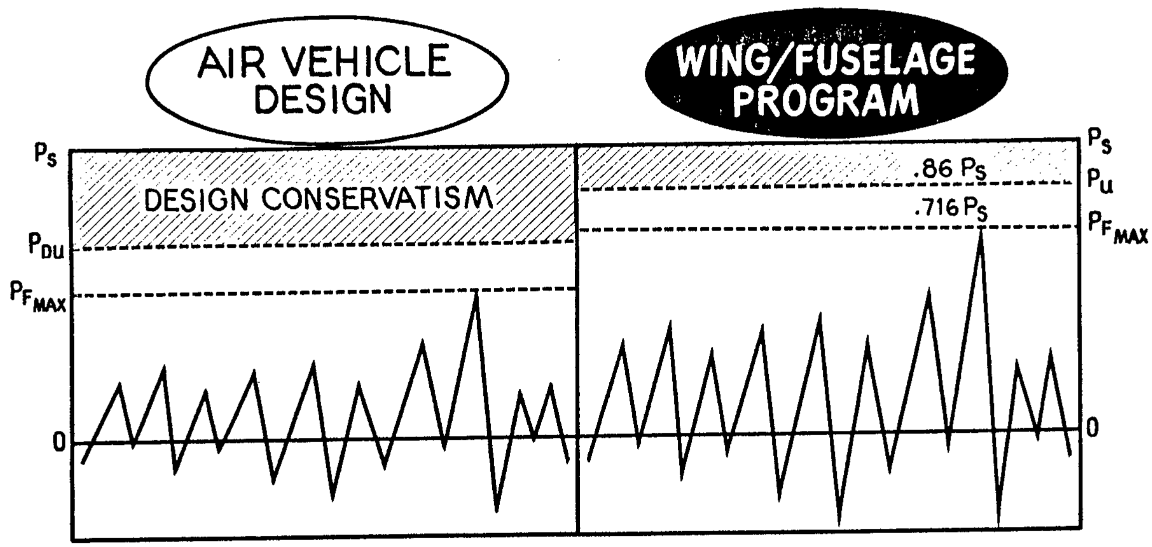
81-12771A
10

ALTERNATE TEST SCHEME No. 2



81-12788

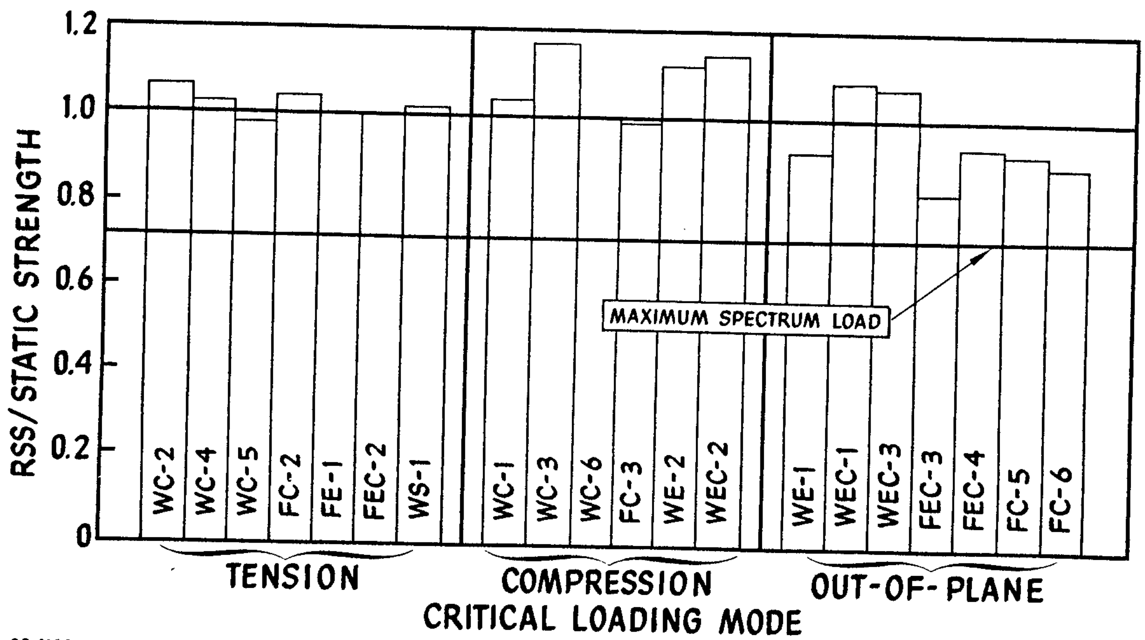
FATIGUE TEST STRAIN LEVELS



P_S = AVERAGE STATIC FAILURE LOAD
P_{DU} = DESIGN ULTIMATE LOAD
P_u = TEST PROGRAM ULTIMATE LOAD
$P_{F_{MAX}}$ = MAXIMUM FATIGUE SPECTRA LOAD

80-12645
10

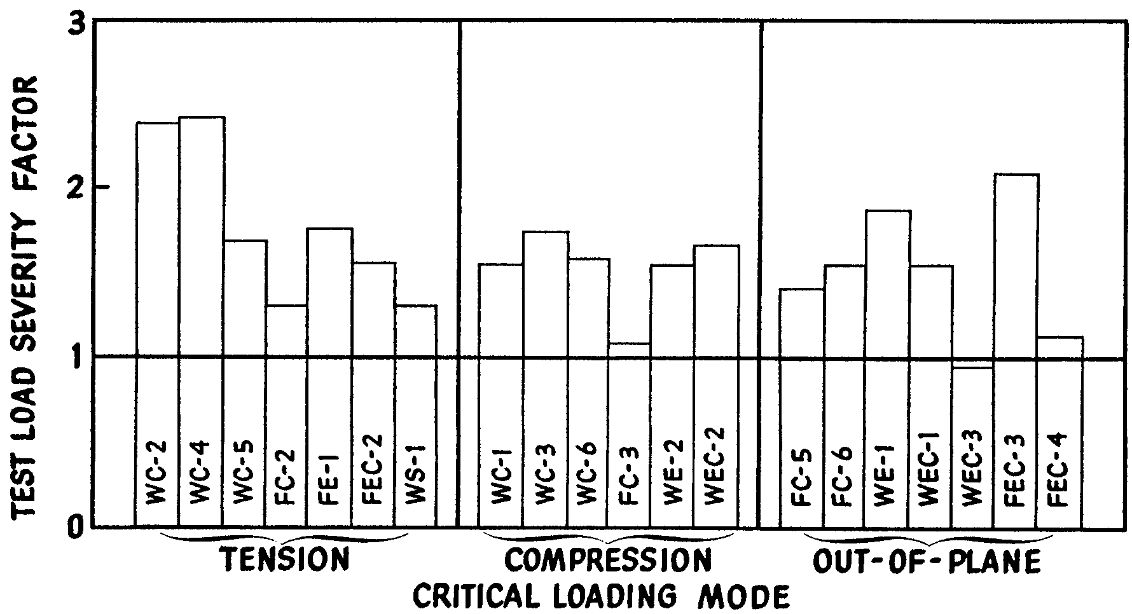
RT/AMBIENT FATIGUE RSS DATA



83-11007
10

RT/AMBIENT FATIGUE

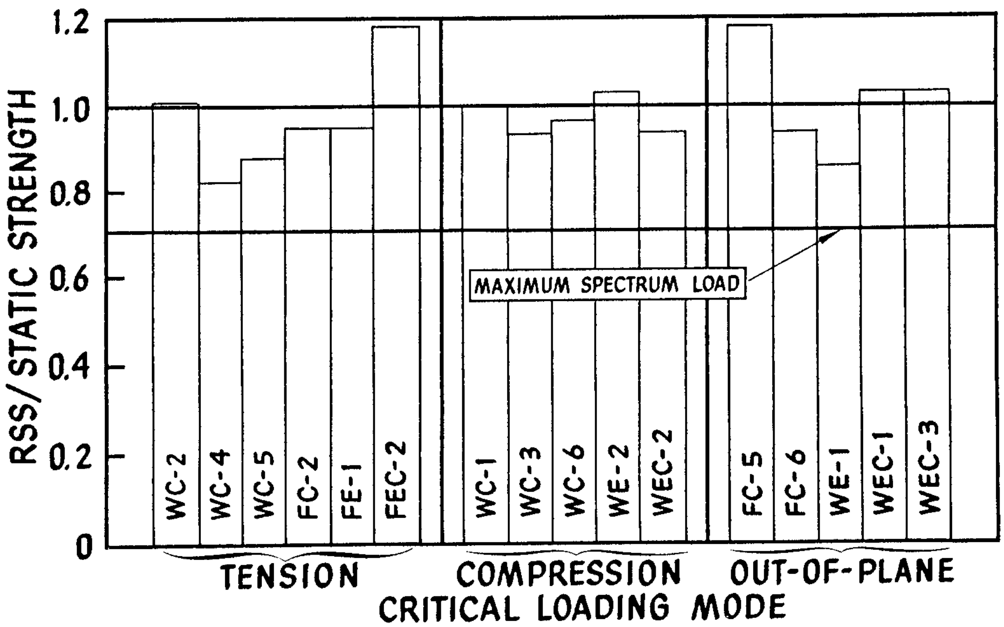
TEST LOAD SEVERITY FACTORS



83-11008
10

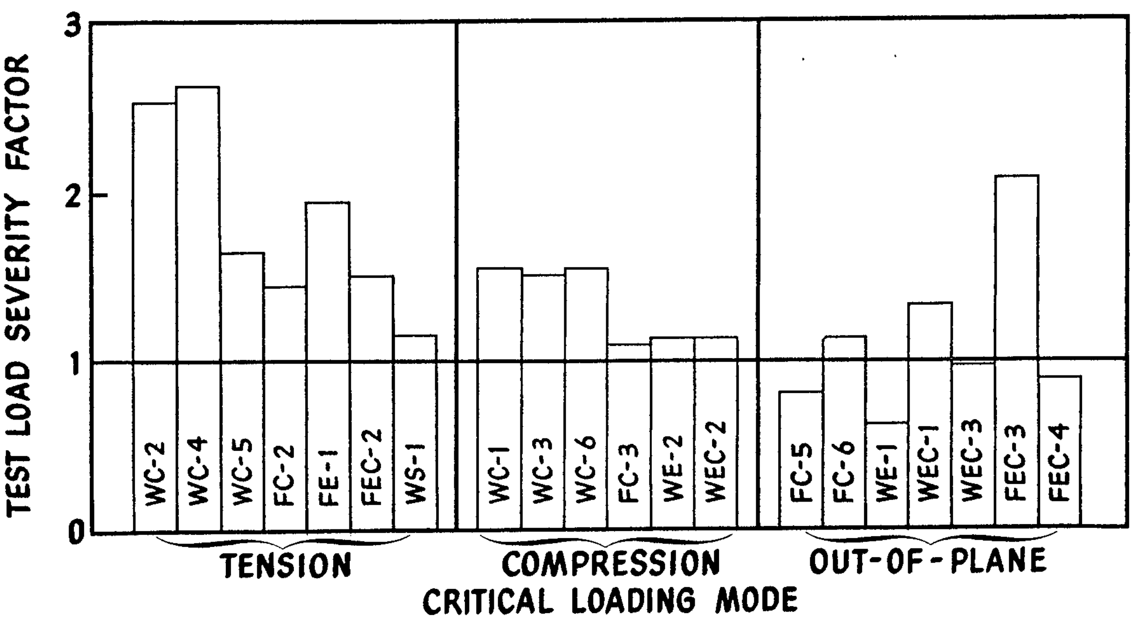
BASELINE ACCELERATED FATIGUE RSS DATA

250 F/WET



ENVIRONMENTAL FATIGUE

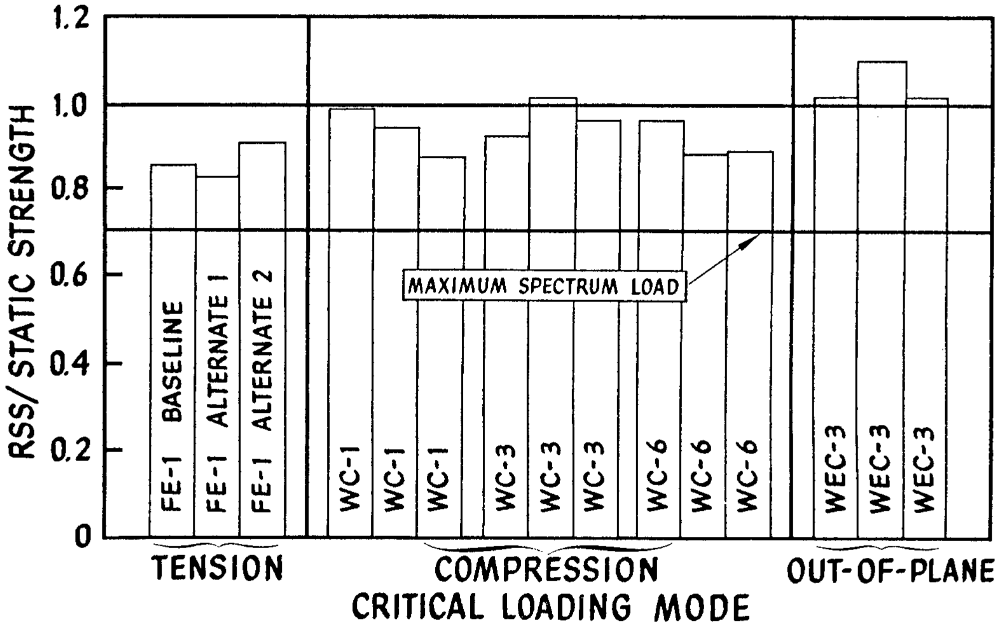
TEST LOAD SEVERITY FACTORS



83-11011
10

COMPARISON OF ENVIRONMENTAL FATIGUE TEST SCHEMES

250 F/WET RSS



CERTIFICATION RECOMMENDATIONS

- COMPOSITE MATERIAL SELECTION SHOULD BE RELATED TO THE AIRCRAFT HYGROTHERMAL ENVIRONMENT
  - NO FATIGUE DESIGN KNOCKDOWN FACTOR
  - RT/AMBIENT TESTING FOR IN-PLANE FAILURE MODES
  - ENVIRONMENTAL TESTING MAY BE NECESSARY FOR OUT-OF-PLANE FAILURE MODES
- IF COMPOSITES ARE USED IN SERVICE ENVIRONMENTS CLOSE TO THEIR Tg
  - FATIGUE DESIGN KNOCKDOWN FACTORS MAY BE NECESSARY
  - COMPLEX ENVIRONMENTAL SIMULATION MAY BE NECESSARY FOR CERTIFICATION TESTING

83-11014
10

DAMAGE TOLERANCE CHARACTERISTICS OF KEVLAR-EPOXY LAMINATES LOADED IN COMPRESSION

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JAMES H. STARNES, JR.

NASA LANGLEY RESEARCH CENTER

W. ALLEN WATERS

KENTRON TECHNICAL CENTER

NINTH ANNUAL MECHANICS OF COMPOSITES REVIEW

DAYTON, OHIO

OCTOBER 24-26, 1983

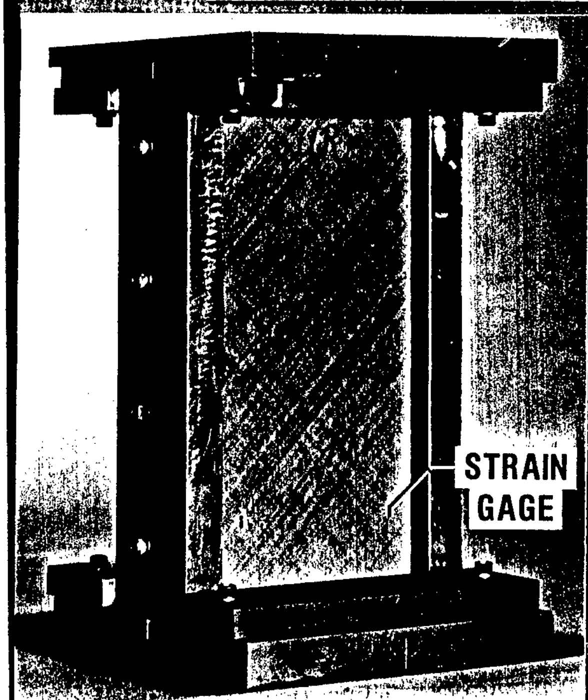
RESEARCH OBJECTIVES

- TO ASSESS THE TOLERANCE OF COMPRESSION-LOADED KEVLAR-EXPOXY LAMINATES TO IMPACT-DAMAGE AND OPEN-HOLE DISCONTINUITIES
- TO DEVELOP STRUCTURAL CONCEPTS WHICH UTILIZE DAMAGE-TOLERANT PROPERTIES CHARACTERISTIC OF KEVLAR-EPOXY LAMINATES
- TO UNDERSTAND THE MECHANISMS OF FAILURE FOR KEVLAR-EPOXY COMPRESSION-LOADED STRUCTURES
- TO ESTABLISH THE STRUCTURAL EFFICIENCY OF PROMISING KEVLAR-EPOXY STRUCTURAL CONCEPTS

CONCLUSIONS

- 45-90 FAMILY KEVLAR-EPOXY LAMINATES ARE VERY TOLERANT TO LOCAL DAMAGE AND DISCONTINUITIES
- KEVLAR-EPOXY LAMINATE COMPRESSION FAILURE MODE APPEARS TO RELIEVE LOCAL STRESS CONCENTRATIONS
- STIFFENED PANEL WITH KEVLAR-EPOXY 45-90 SKIN AND GRAPHITE-EPOXY STIFFENERS CAN PROVIDE 30 PERCENT WEIGHT SAVINGS COMPARED TO ALUMINUM WING DESIGNS
- MECHANICAL ATTACHMENT OF STIFFENERS BENEFICIAL IN PREVENTING DAMAGE PROPAGATION

TEST SPECIMENS AND FIXTURE

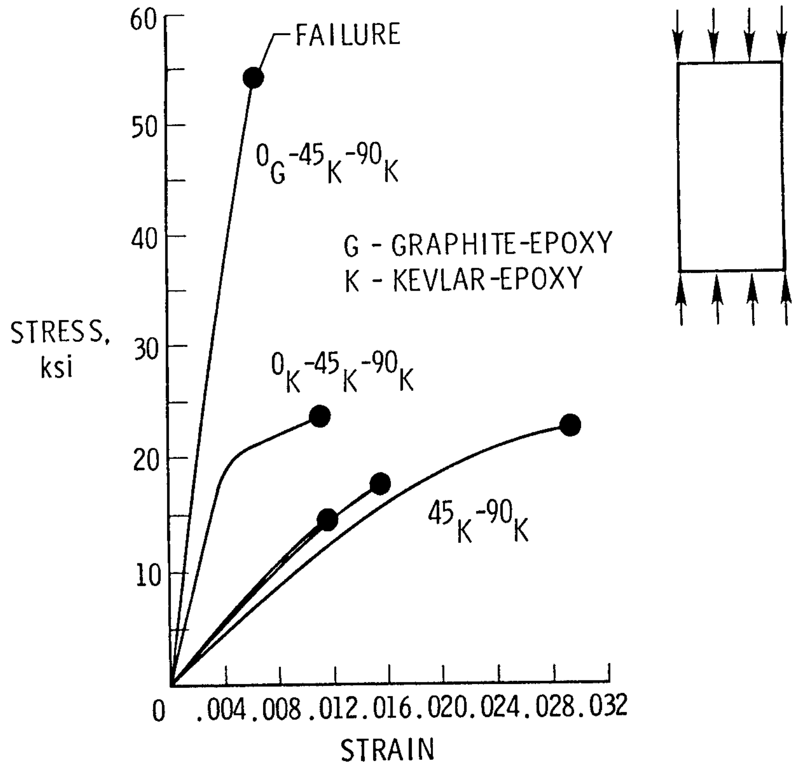


MATERIALS
- KEVLAR 49-5208
- KEVLAR 49-SP328
- T300-5208

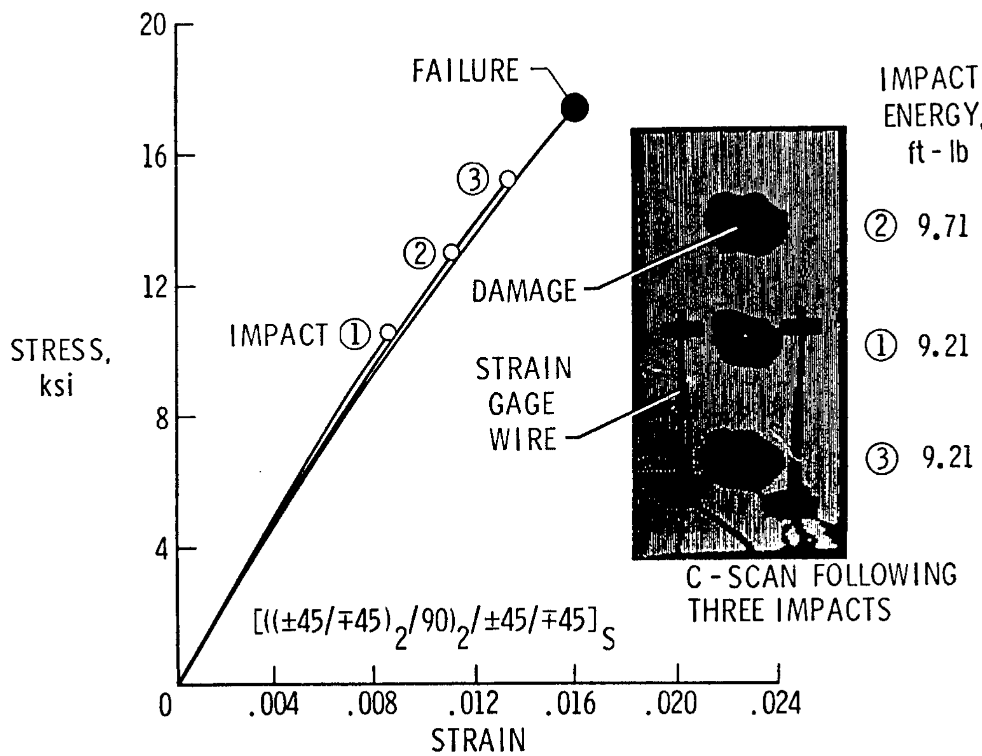
LAMINATES
- 0-45-90 FAMILY
- 45-90 FAMILY

SPECIMEN
- 10-inch LONG
- 5-inch WIDE
- 0.15 TO 0.40-inch THICK

LAMINATE STRESS-STRAIN RESPONSES

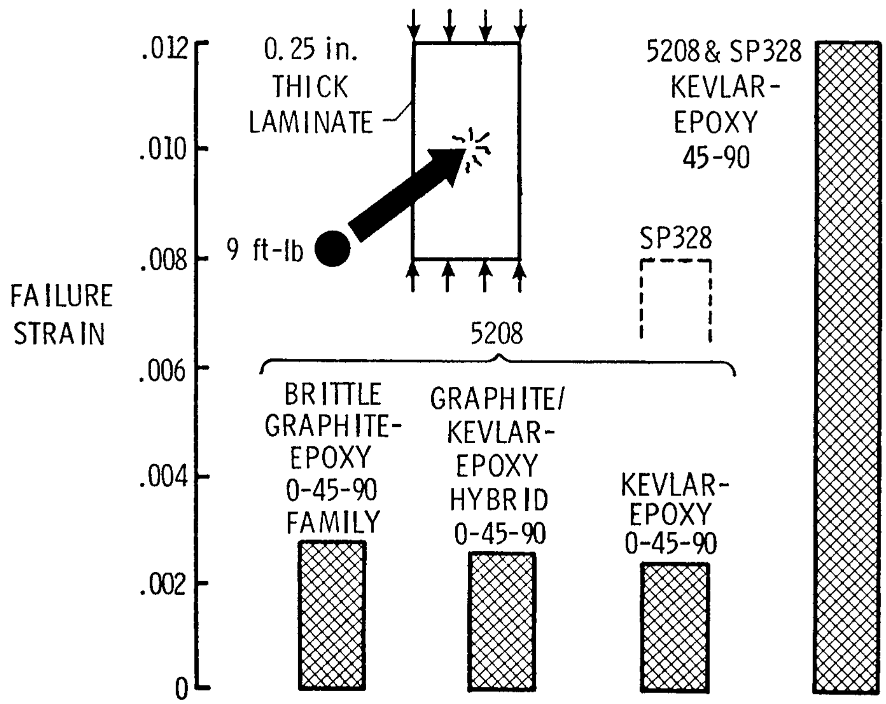


KEVLAR-EPOXY SPECIMEN SUBJECTED TO MULTIPLE-IMPACT

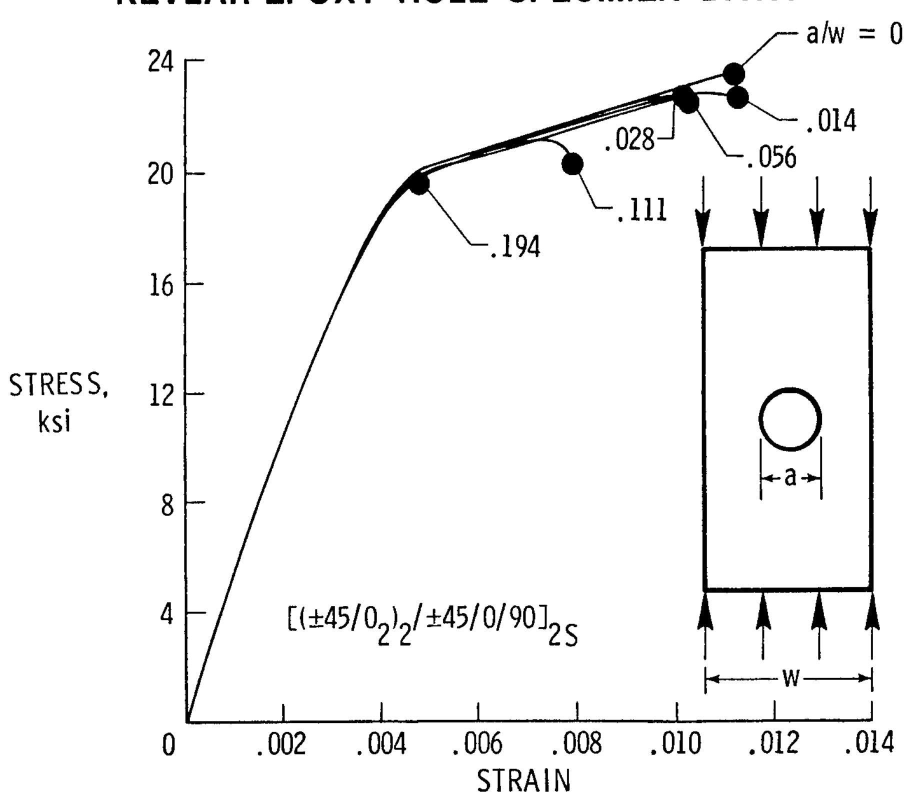


C-SCAN FOLLOWING THREE IMPACTS

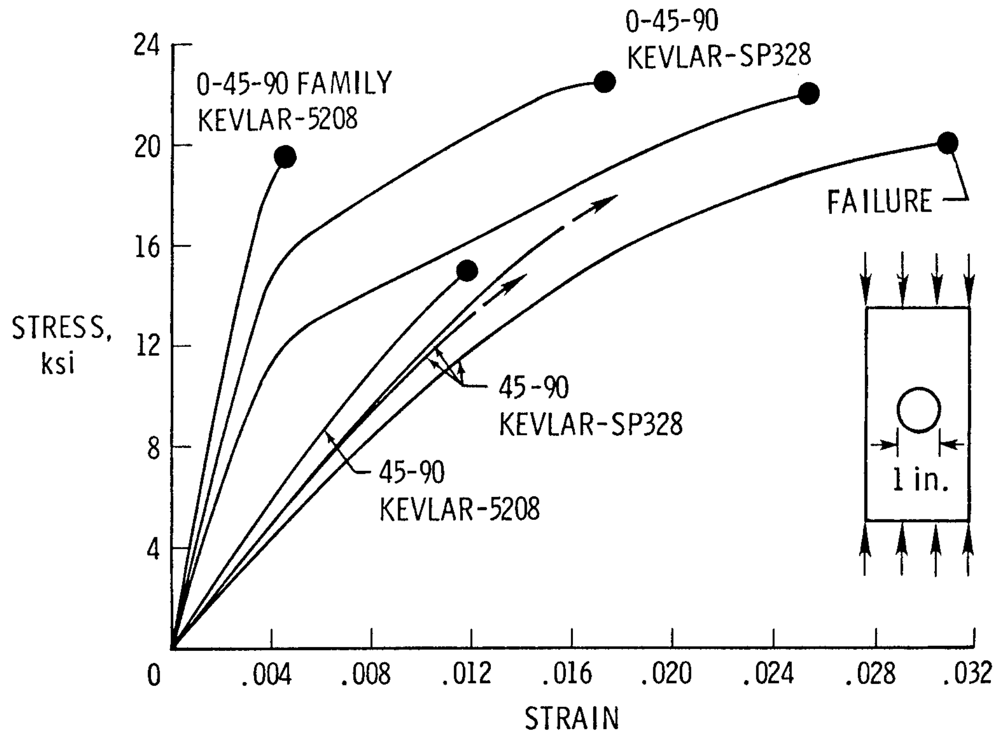
IMPACT DAMAGE TOLERANCE COMPARISON



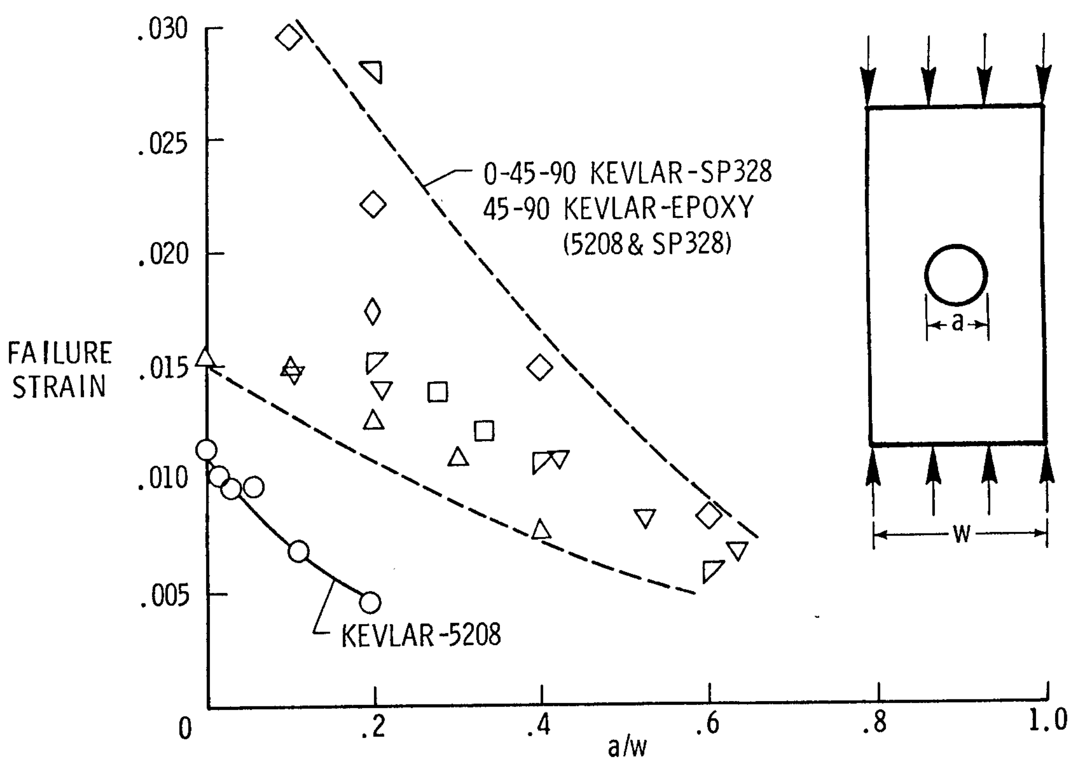
KEVLAR-EPOXY HOLE SPECIMEN DATA

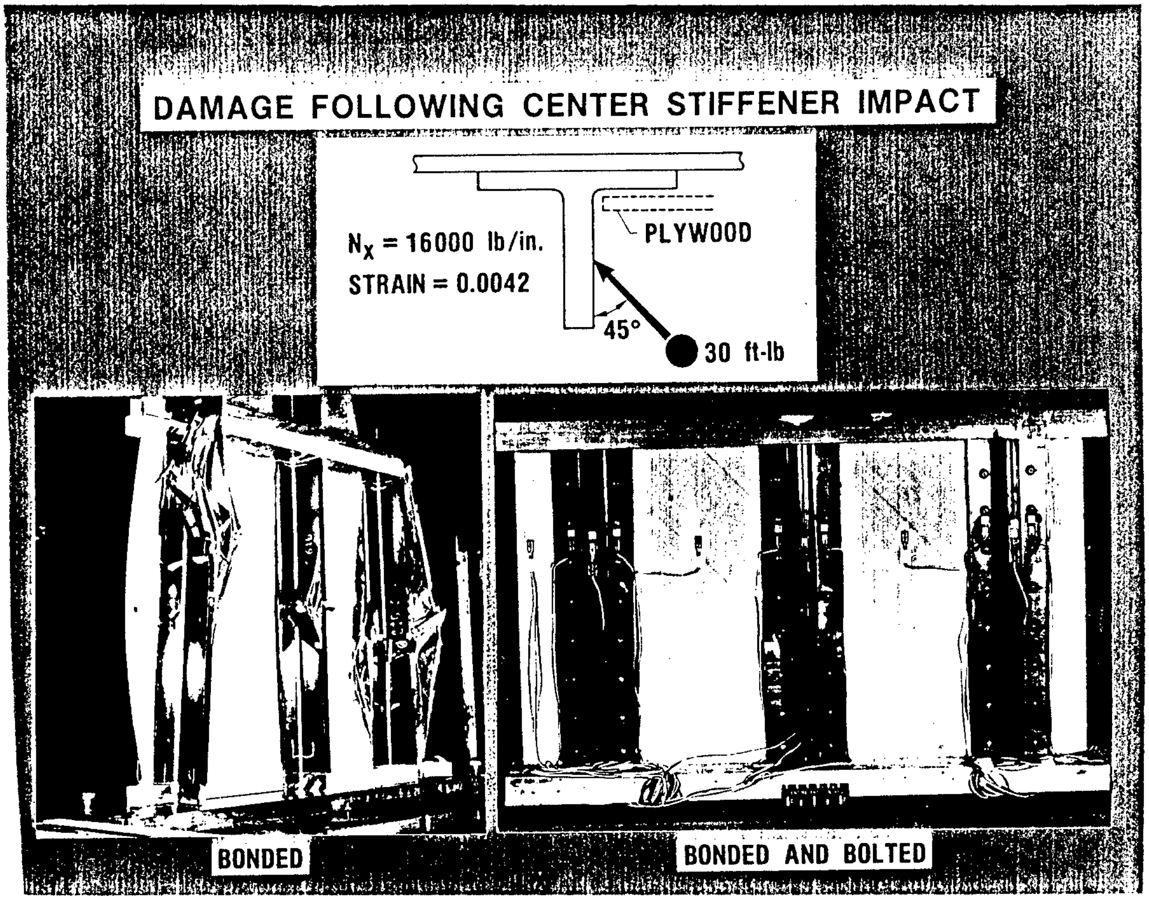


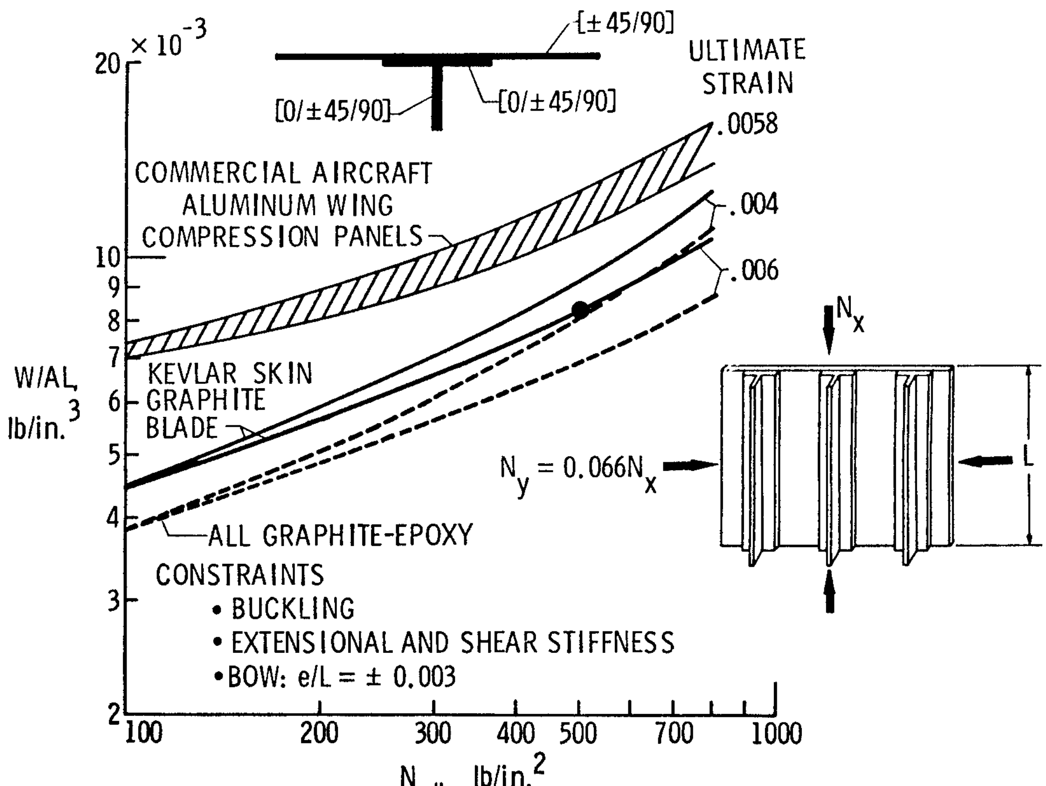
STRESS-STRAIN RESPONSE WITH 1-INCH HOLE



KEVLAR-EPOXY HOLE SPECIMEN FAILURE STRAIN




DAMAGE FOLLOWING CENTER STIFFENER IMPACT
N_X = 16000 lb/in.
STRAIN = 0.0042
PLYWOOD
45°
30 ft-lb
BONDED
BONDED AND BOLTED


STRUCTURAL EFFICIENCY COMPARISON
× 10^-3
[±45/90]
[0/±45/90]
[0/±45/90]
ULTIMATE
STRAIN
.0058
.004
.006
COMMERCIAL AIRCRAFT
ALUMINUM WING
COMPRESSION PANELS
KEVLAR SKIN
GRAPHITE
BLADE
ALL GRAPHITE-EPOXY
W/AL, lb/in.^3
N_X
N_y = 0.066N_X
L
CONSTRAINTS
• BUCKLING
• EXTENSIONAL AND SHEAR STIFFNESS
• BOW: e/L = ± 0.003
N_X/L, lb/in.^2
20
10
9
8
7
6
5
4
3
2
100
200
300
400
500
700
1000

COMPRESSION STRENGTH OF COMPOSITES WITH EMBEDDED DELAMINATIONS

R. B. DEO
R. S. WHITEHEAD
M. M. RATWANI

NORTHROP CORPORATION, AIRCRAFT DIVISION

AFWAL CONTRACT F33615-82-C-3213	-	"DAMAGE TOLERANCE OF COMPOSITES"
AFWAL PROJECT ENGINEER	-	E. DEMUTS
PROGRAM MANAGERS:		
BOEING		J. McCARTY
NORTHROP		M. RATWANI

PROGRAM OBJECTIVE

- ● DEVELOP DAMAGE TOLERANCE REQUIREMENTS FOR ADVANCED COMPOSITE STRUCTURES UTILIZED IN USAF AIRCRAFT
- ○ VALIDATE ADEQUACY OF REQUIREMENTS
- ○ DEVELOP AND DEMONSTRATE METHODS OF DESIGN COMPLIANCE
- ○ ENHANCE COMPOSITE DAMAGE TOLERANCE CAPABILITY

PROGRAM CONTENT

TASK I - TECHNOLOGY BASE DEVELOPMENT

- ASSESS AVAILABLE TECHNOLOGY
- EXTEND DATA BASE
- DEFINE D/T DRAFT REQUIREMENTS

TASK II - COMPONENT DESIGN

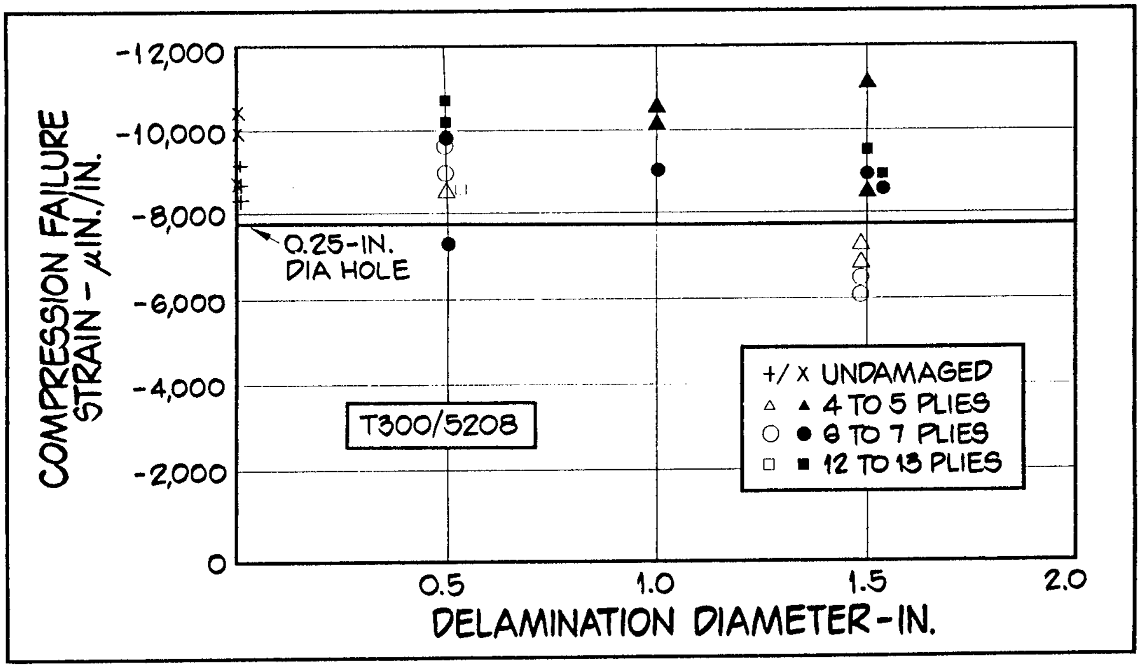
TASK III - DAMAGE TOLERANCE QUALIFICATION

TASK IV - TECHNOLOGY CONSOLIDATION

TECHNOLOGY BASE DEVELOPMENT OBJECTIVES

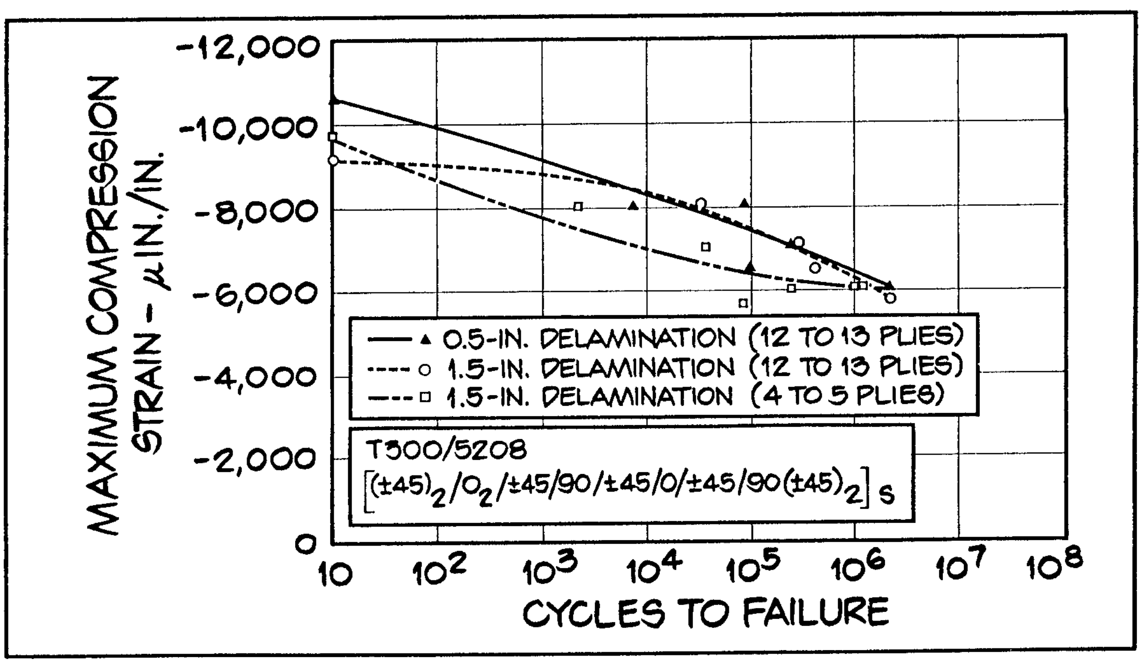
- DETERMINE INFLUENCE OF COMMONLY OCCURRING MANUFACTURING FLAWS AND IN-SERVICE DAMAGE ON STATIC AND FATIGUE RESPONSE OF COMPOSITES
- IDENTIFY DATA GAPS
- EXPAND DATA BASE TO ENABLE EVALUATION OF FLAW/DAMAGE CRITICALITY AND VERIFICATION OF ANALYSIS METHODOLOGY

COMPRESSION STATIC STRENGTH



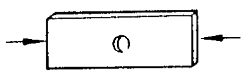
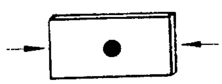
83-13321
10E

COMPRESSION-COMPRESSION FATIGUE



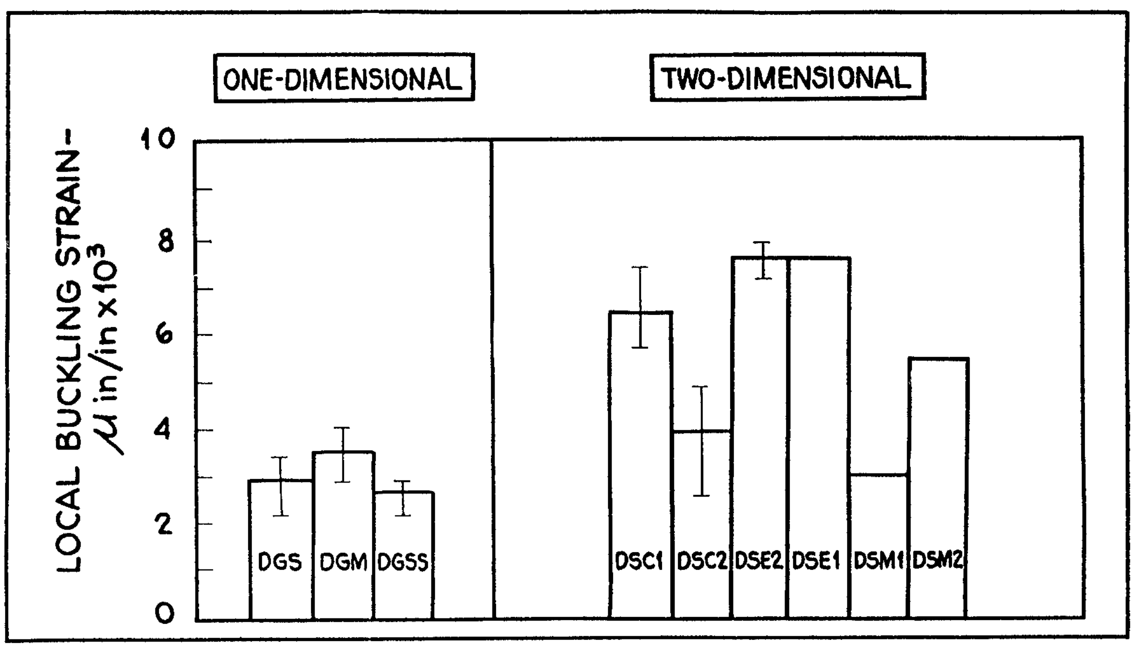
83-13322
10E

BASELINE MATERIAL TEST MATRIX

SPECIMEN	DEFECT DAMAGE	TEST ENVIRONMENT	NUMBER OF SPECIMENS STATIC	NUMBER OF SPECIMENS FATIGUE
	HOLE FLAWS	RTD ETW	9 6	15 6
	1-D DELAMINATION	RTD ETW	12 6	18 6
	2-D DELAMINATION	RTD ETW	21 12	27 12
	2-D DELAMINATION	RTD	6	12
	2-D DELAMINATION	RTD	6	12

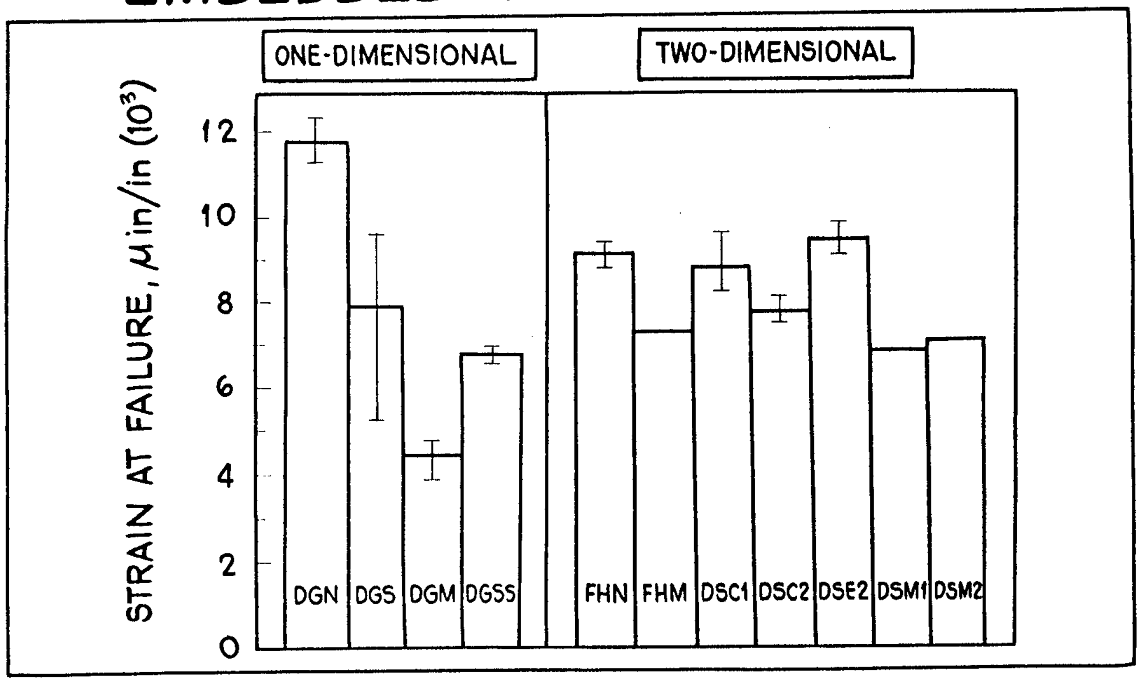
83-13364
10E

INITIAL BUCKLING STRAINS FOR EMBEDDED DELAMINATIONS



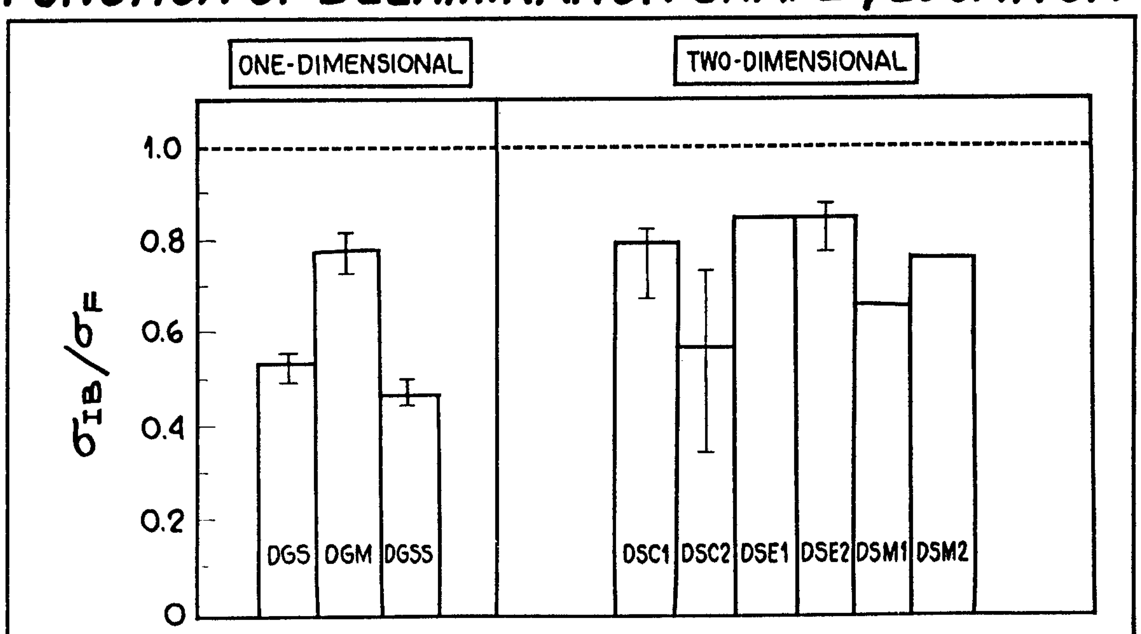
83-13380
10E

FAILURE STRAINS FOR EMBEDDED DELAMINATIONS



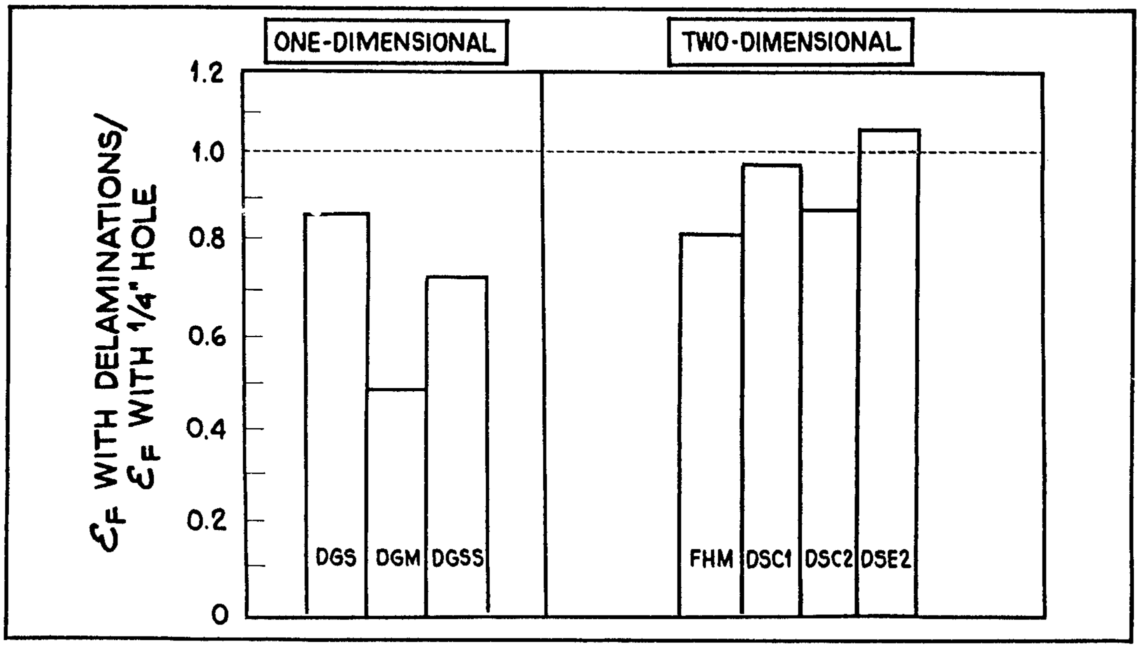
83-13381
10E

NORMALIZED INITIAL BUCKLING LOAD AS A FUNCTION OF DELAMINATION SHAPE & LOCATION



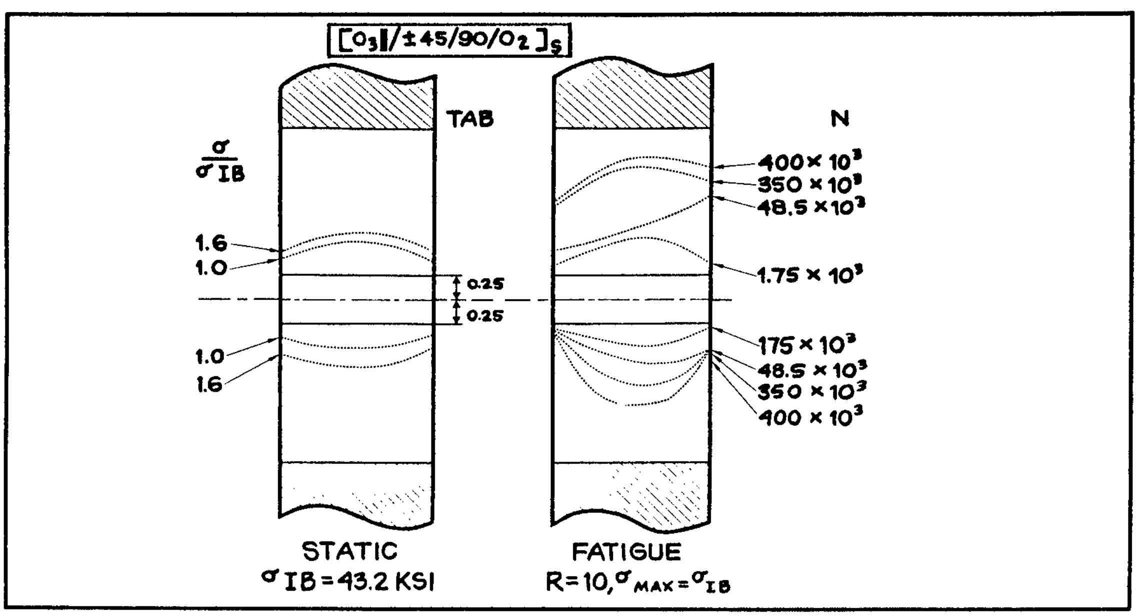
83-13382
10E

STRENGTH OF LAMINATES WITH EMBEDDED DELAMINATIONS



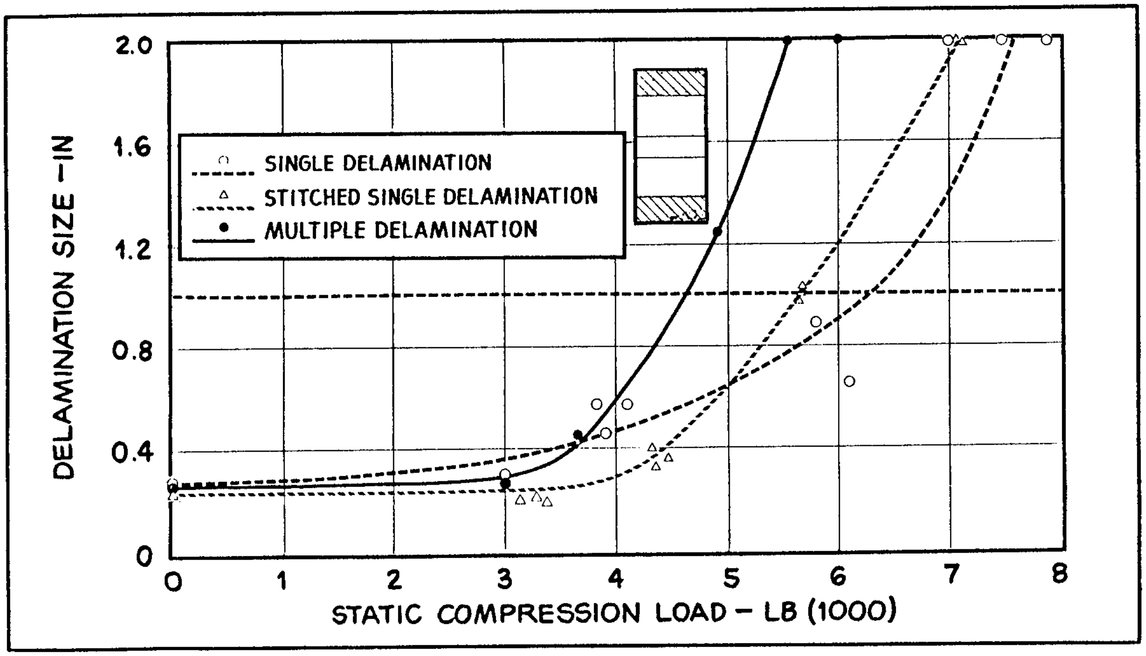
83-13383
10E

1-D DELAMINATION GROWTH UNDER STATIC & FATIGUE LOADING



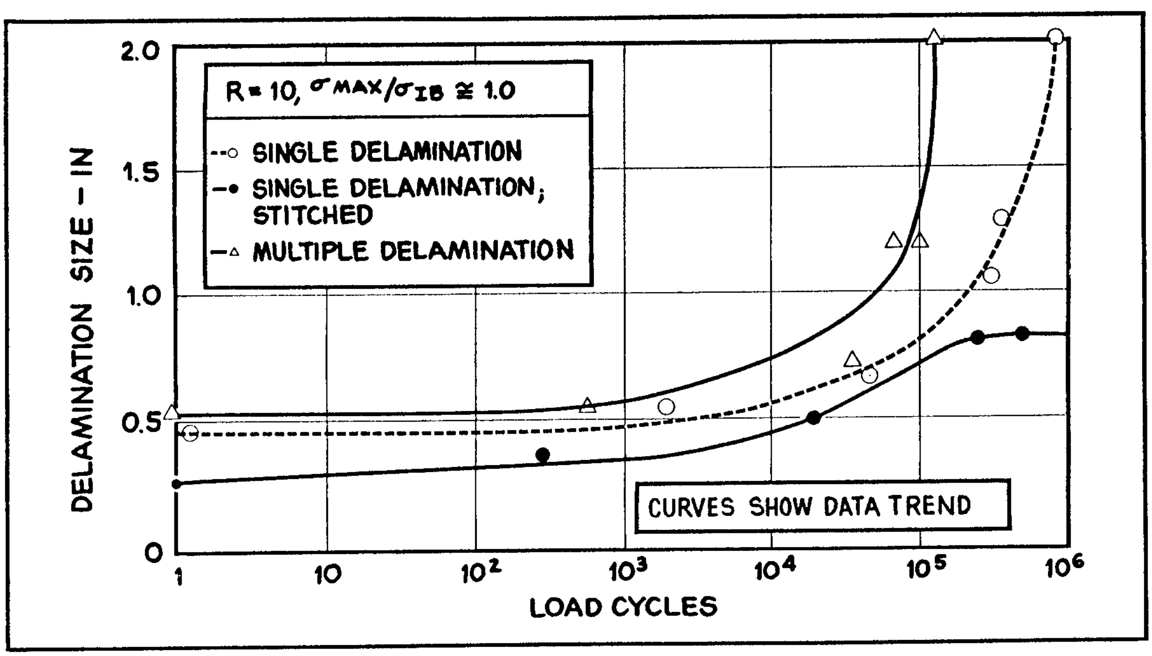
83-13368
10E

1-D DELAMINATION GROWTH UNDER STATIC LOADING



83-13369
10E

DELAMINATION GROWTH RATE COMPARISON



83-13373
10E

CONCLUSIONS

- ONE-DIMENSIONAL DELAMINATION GROWTH UNDER STATIC LOAD NOT INFLUENCED BY STITCHING
- STITCHING EFFECTIVE IN RETARDING ONE-DIMENSIONAL DELAMINATION GROWTH UNDER FATIGUE LOADING
- SCATTER IN FATIGUE GROWTH RATES OF ONE-DIMENSIONAL DELAMINATIONS ATTRIBUTABLE TO SCATTER IN INITIAL BUCKLING LOADS
- STATIC RESPONSE OF SMALL, NEAR SURFACE AND LARGE, DEEP 2-D DELAMINATIONS SIMILAR
- INITIAL BUCKLING LOAD FOR LARGE, NEAR SURFACE 2-D DELAMINATION APPROXIMATELY 50% OF STATIC STRENGTH
- STATIC RESPONSE OF MULTIPLE 2-D DELAMINATIONS SIMILAR TO THAT FOR DELAMINATION CLOSEST TO THE SURFACE

83-13378
11D, 10E

FRACTURE TOUGHNESS OF COMPOSITE LAMINATES

C. C. POE, JR.
NASA LANGLEY RESEARCH CENTER

OCTOBER 24-26, 1983

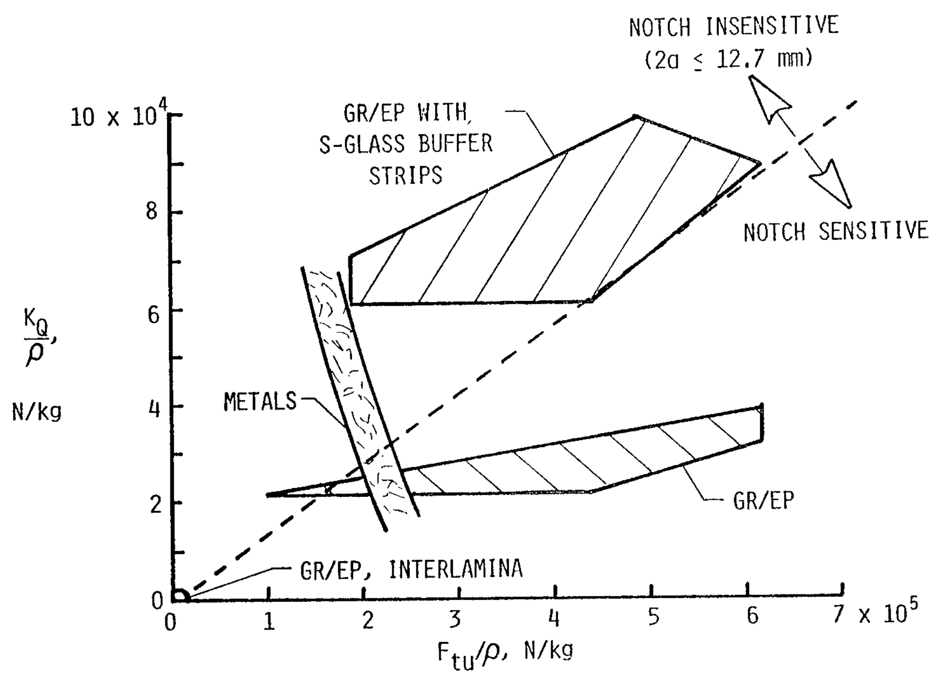
OBJECTIVES

- PREDICT FRACTURE TOUGHNESS OF LAMINATES FROM FIBER AND MATRIX PROPERTIES.
- DETERMINE WHAT MAKES COMPOSITES TOUGH.

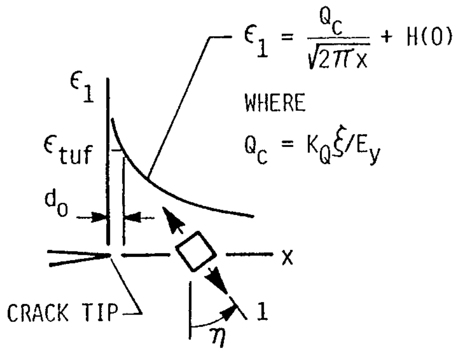
CONCLUSIONS

- STRONG 0° FIBERS AND STIFF NON-0° FIBERS MAKE COMPOSITES TOUGH.
- WEAK MATRICES CAN MAKE COMPOSITES TOUGH.
- THICK LAMINATES CAN BE LESS TOUGH.

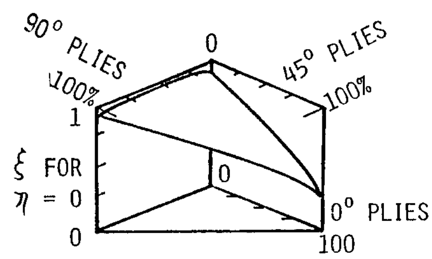
FRACTURE TOUGHNESS AND ULTIMATE TENSILE STRENGTH



PREDICTING FRACTURE TOUGHNESS

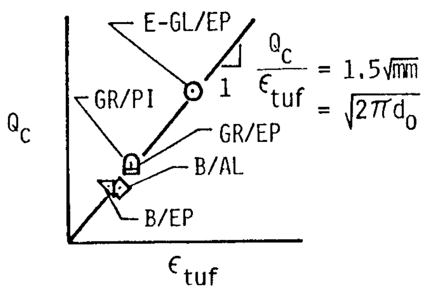


$[0_i/45_j/90_k]_S$ GR/EP



$$\xi = \left(1 - \nu_{yx}\sqrt{E_x/E_y}\right)\left(\sqrt{E_y/E_x}\ \sin^2 \eta + \cos^2 \eta\right)$$

η = PRINCIPAL FIBER DIRECTION



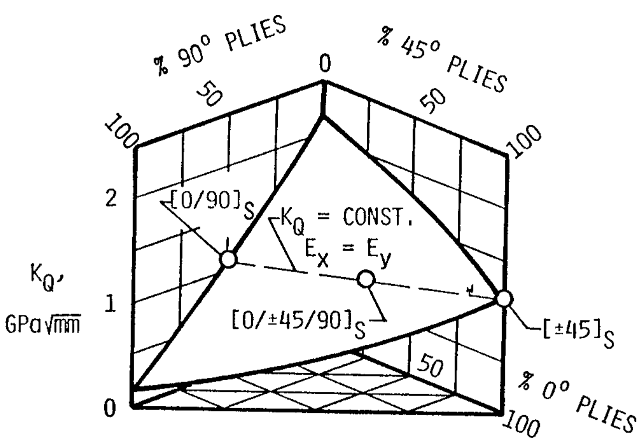
FRACTURE TOUGHNESS IS PREDICTED BY

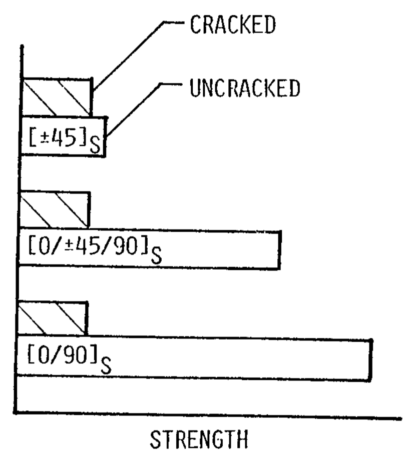
$$K_Q = Q_c E_y / \xi$$
$$= 1.5 \epsilon_{tuf} E_y / \xi$$

INFLUENCE OF LAYUP ON TOUGHNESS

T300/5208, $[0_i/\pm45_j/90_k]_S$

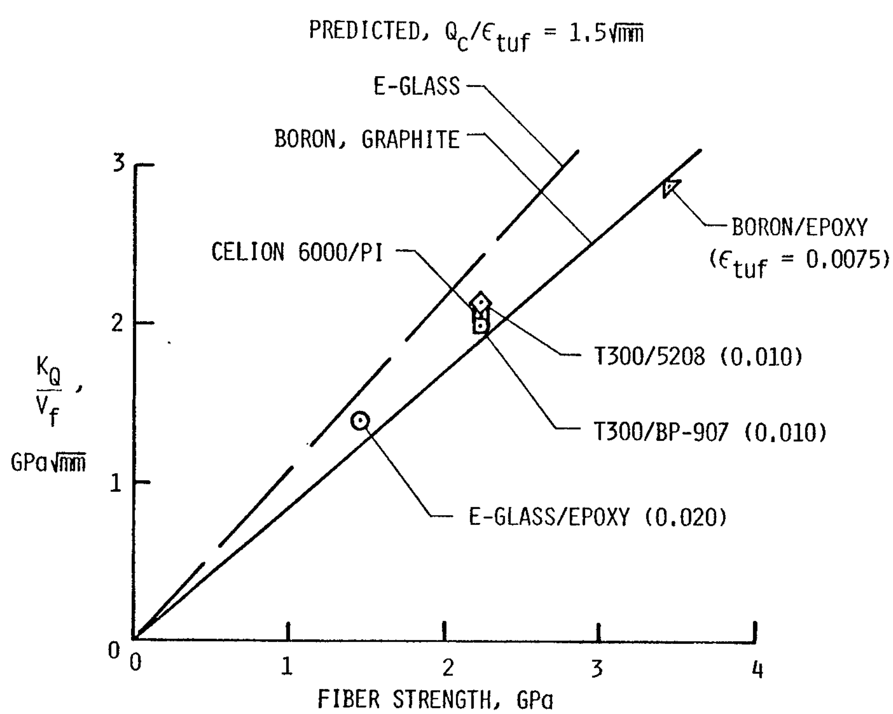
$Q_c/\epsilon_{tuf} = 1.5\sqrt{mm}$





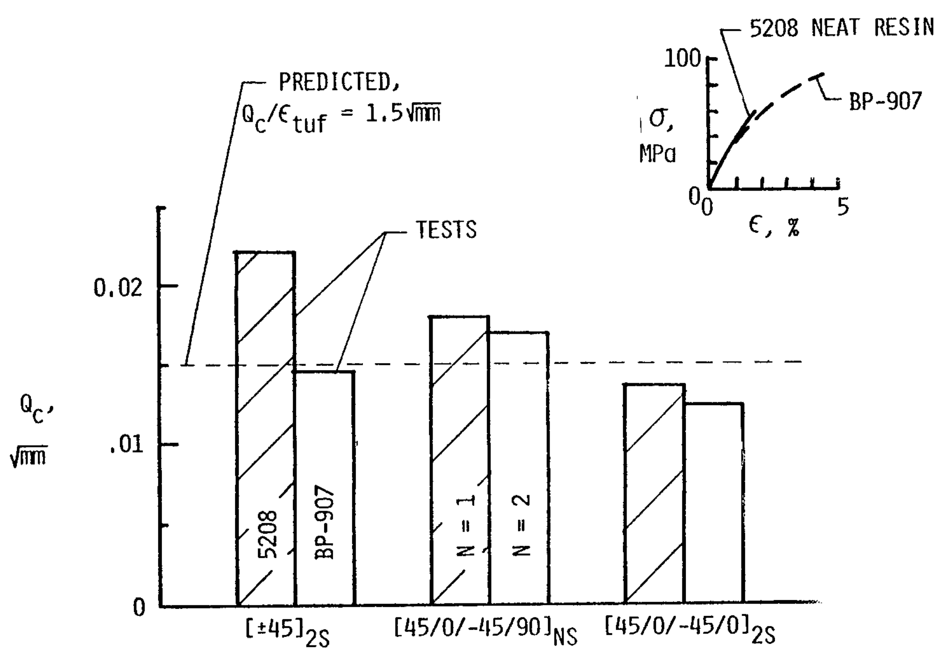
INFLUENCE OF FIBER STRENGTH ON TOUGHNESS

$[0/\pm45/90]_{NS}$



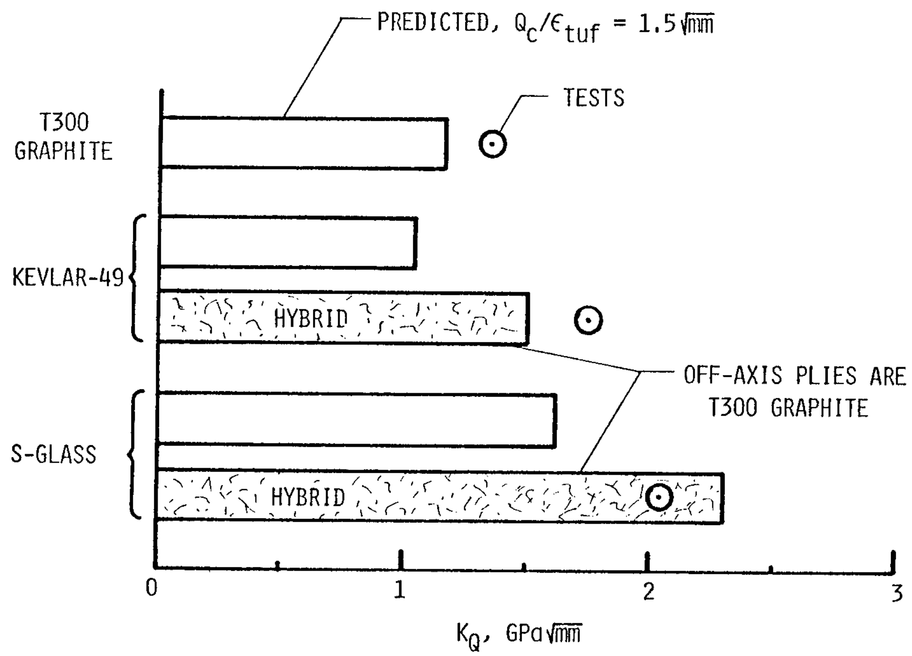
INFLUENCE OF MATRIX ON TOUGHNESS

T300/5208 AND T300/BP-907



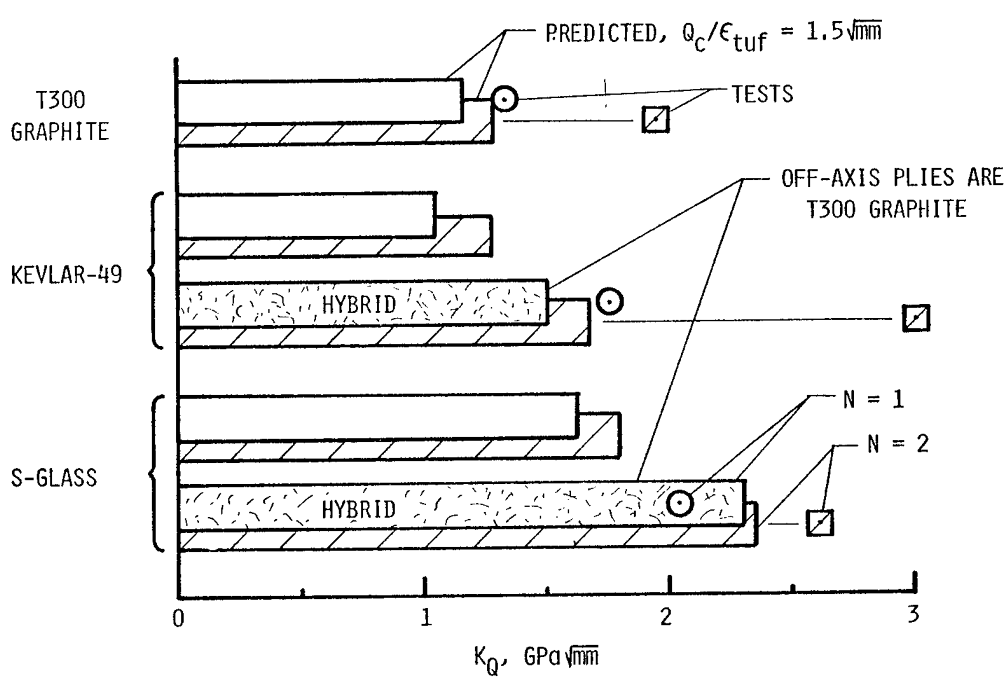
TOUGHNESS OF HYBRIDS

$[45/0/-45/90]_{2S}$, 5208 MATRIX



TOUGHNESS OF HYBRIDS - VARIOUS NUMBER OF 0^o PLIES

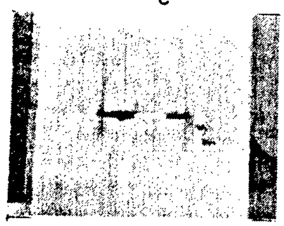
$[45/0_N/-45/90]_{2S}$, 5208 MATRIX



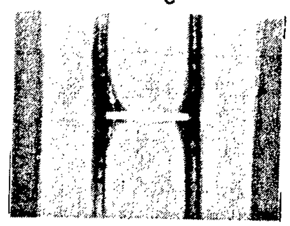
RADIOGRAPHS OF 50-mm-WIDE SPECIMENS

UNLESS OTHERWISE NOTED, $2a = 15$ mm

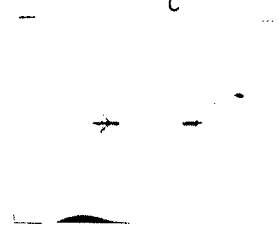
$[90/0]_{2S}$
T300/5208
$0.91S_C$



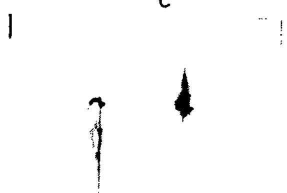
$[0_2/90/0]_S$
T300/5208
$0.95S_C$



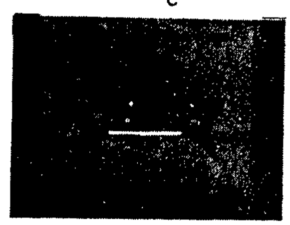
$[(90/0)_2/45/0/-45/0]_S$
T300/5208
$0.99S_C$



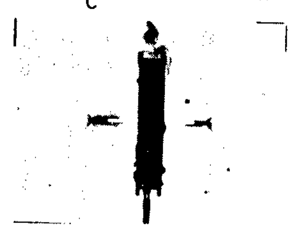
$[0_2/45/0_2/-45/0_2]_S$
T300/5208
$0.95S_C$



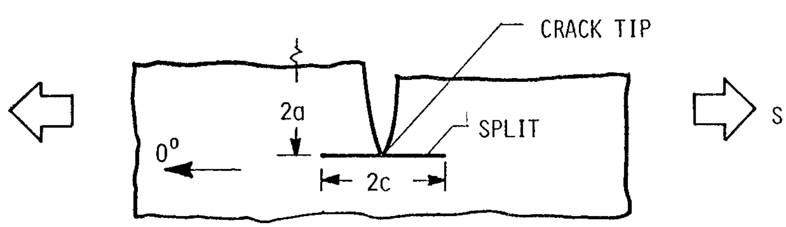
$[45/0/-45/90]_S$
T300/5208
$0.95S_C$

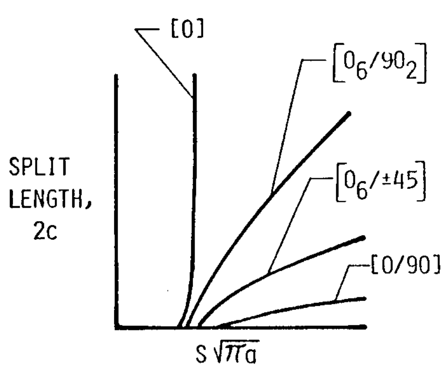


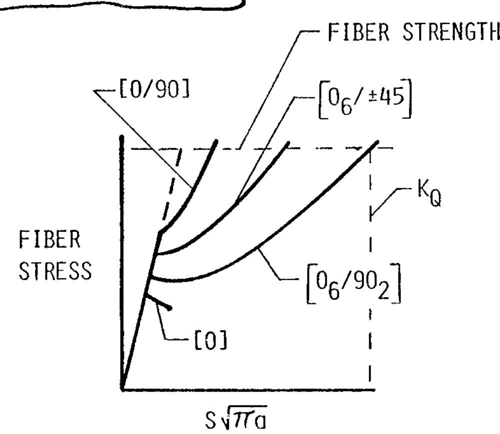
$[45/0/-45/90]_S$
T300/BP-907
$0.94S_C$ ($2a = 20$ mm)



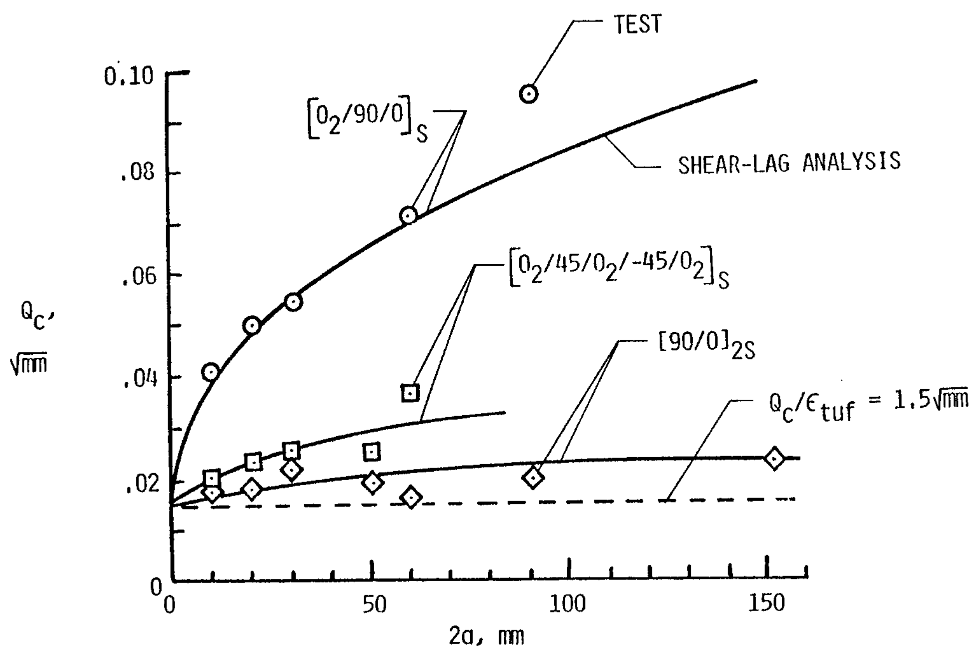
SHEAR-LAG ANALYSIS OF CRACK-TIP SPLITTING





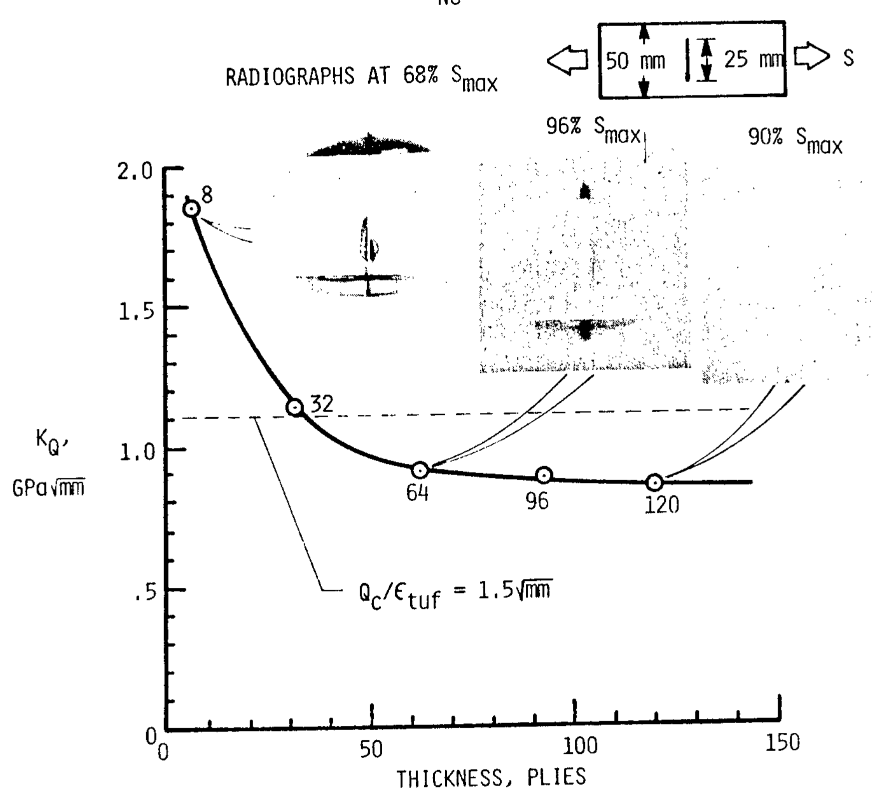


TOUGHNESS PREDICTED BY SHEAR-LAG ANALYSIS



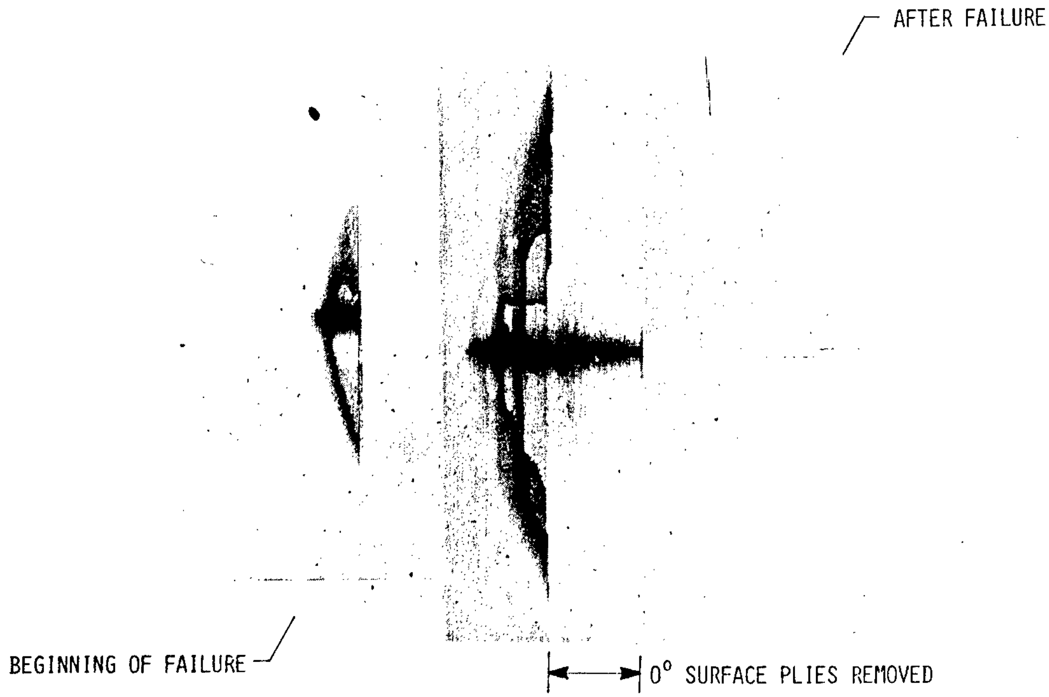
TOUGHNESS, CRACK-TIP DAMAGE, AND THICKNESS

$[0/90]_{NS}$ T300/5208



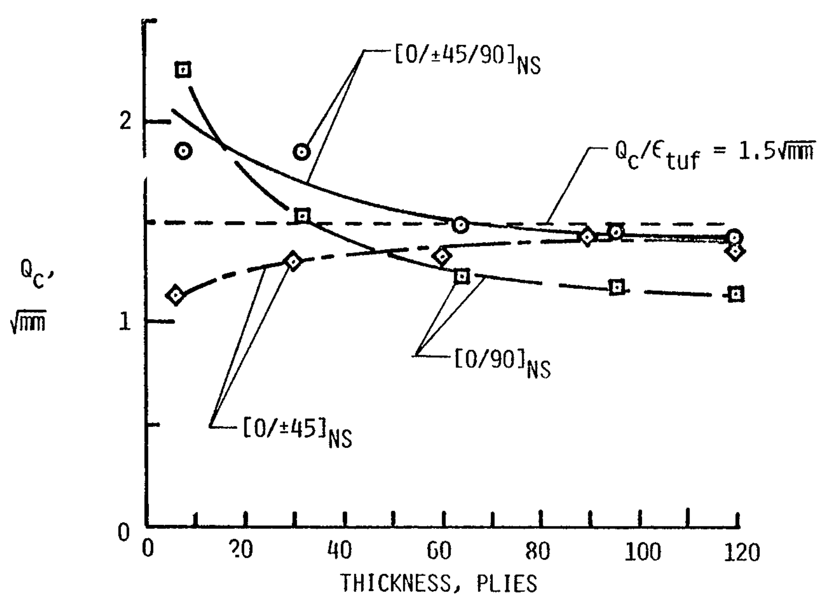
EVIDENCE OF SURFACE DAMAGE

RADIOGRAPHS OF COMPACT $[0/90]_{24S}$ T300/5208 SPECIMEN



TOUGHNESS AND THICKNESS

T300/5208



PROGRESSIVE FRACTURE OF COMPOSITES

BY

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NASA LEWIS RESEARCH CENTER
CLEVELAND, OHIO 44135

MECHANICS OF COMPOSITES REVIEW
OCTOBER 24 - 26, 1983
DAYTON, OHIO

OBJECTIVE

TO DEVELOP AND REFINE MODELS/PROCEDURES FOR PREDICTING PROGRESSIVE COMPOSITE FRACTURE INCLUDING CHARACTERIZATION OF CRACK PROPAGATION AND FAILURE MODES.

CONCLUSIONS

- THE COMPOSITE DURABILITY STRUCTURAL ANALYSIS (CODSTRAN) COMPUTER CODE AND THE REAL-TIME ULTRASONIC C-SCAN (RUSCAN) EXPERIMENTAL FACILITY ARE EFFECTIVE METHODS OF STUDYING PROGRESSIVE FRACTURE OF COMPOSITES.

- CODSTRAN GIVES THE INVESTIGATOR THE CAPABILITY TO PREDICT FAILURE STRESS AND FRACTURE PROPAGATION PATTERNS.

- RUSCAN SENSITIVITY ALLOWS ACCURATE TRACKING OF CRACK INITIATION AND PROGRESSIVE FRACTURE.

- FRACTURE PATTERNS AND CRACK OPENING DISPLACEMENTS OBSERVED VIA RUSCAN AND PREDICTED BY CODSTRAN ARE IN GOOD AGREEMENT.

BACKGROUND/APPROACH

BACKGROUND: EVALUATING COMPOSITE DURABILITY AND STRUCTURAL RELIABILITY IS DEPENDENT UPON THE CAPABILITY TO CHARACTERIZE FLAWS AND PREDICT DAMAGE ACCUMULATION. THIS CAPABILITY IS BASED UPON DETERMINING LOCAL STRESSES IN COMPOSITE LAMINATES AND SUBSEQUENT APPLICATIONS OF FAILURE CRITERIA. TO DATE, PREDICTIVE METHODS HAVE NOT ACCOUNTED FOR DEFECT GROWTH IN COMPOSITES AND PROPAGATION TO FRACTURE.

APPROACH: USE OF UNIQUE LEWIS RESEARCH CENTER CAPABILITIES: COMPOSITE DURABILITY STRUCTURAL ANALYSIS (CODSTRAN), REAL-TIME ULTRASONIC C-SCAN (RUSCAN)

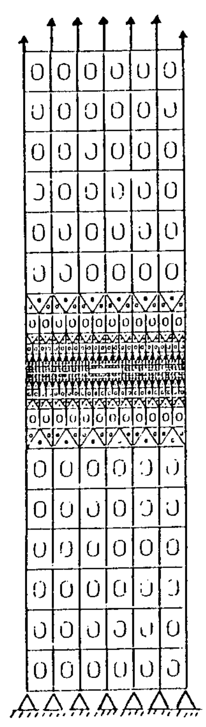
- CODSTRAN - AN UPWARD INTEGRATED MECHANISTIC METHOD ENCOMPASSING COMPOSITE MECHANICS, LAMINIATE THEORY, STRUCTURAL ANALYSIS (FINITE ELEMENT), AND FAILURE CRITERIA.

- RUSCAN - NONDESTRUCTIVE ULTRASONIC TECHNIQUE FOR VERIFICATION OF CODSTRAN PREDICTED RESULTS.

PRESENTATION OUTLINE

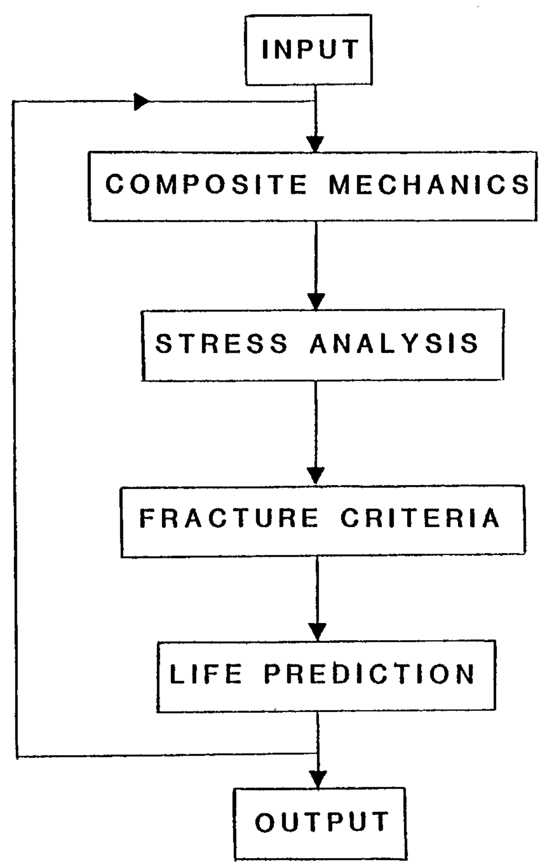
- ANALYTICAL METHODS
  - CODSTRAN - COMPOSITE MECHANICS, FINITE ELEMENT STRESS ANALYSIS, AND FAILURE CRITERIA
- SPECIMEN PREPARATION - FABRICATION, NOTCHING PROCESS
- EXPERIMENTAL CAPABILITY
  - LOAD FRAME
  - RUSCAN
- RESULTS

FINITE ELEMENT MODEL



- 534 ELEMENTS
- 446 NODAL POINTS
- 871 DEGREES OF FREEDOM

CODSTRAN FLOW CHART



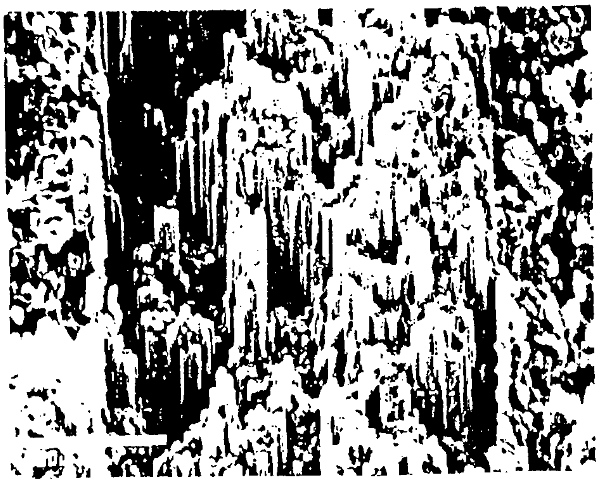
FAILURE CRITERIA AND FRACTURE PROPAGATION

FAILURE CRITERIA

- o PLY COMBINED STRESS FAILURE -- 5 MODES
- o INTERPLY FAILURE -- 1 MODE

FRACTURE CRITERIA

- o PLY LEVEL FRACTURE
- o LAMINATE FRACTURE
- o INTERPLY FRACTURE



TENSILE FAILURE IN 4 PLY [±30] LAMINATE

SHEAR FAILURE IN 4 PLY [±45] LAMINATE

SPECIMEN SCHEMATIC

PLY

- o FIBERITE 934 PREPREG
- o T 300 GRAPHITE FIBER

PANEL

- o 4 PLIES CURED AT 350^0 FOR 2.5 HOURS

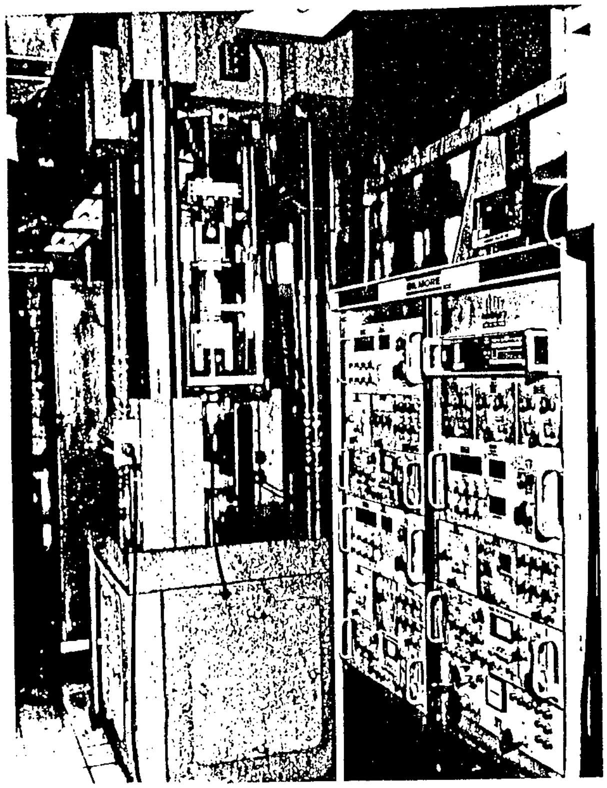
SPECIMEN

- o SPECIMEN DIMENSIONS: 2.0 x .02 INCHES
- o BEVELED ALUMINUM TABS

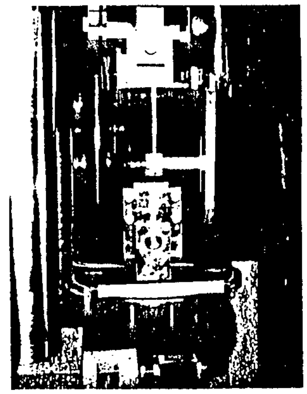
DEFECT/SLIT

- o SLIT DIMENSIONS: 0.25 x 0.05 INCHES
- o NOTCHING BY ULTRASON ABRASIVE SLURRY

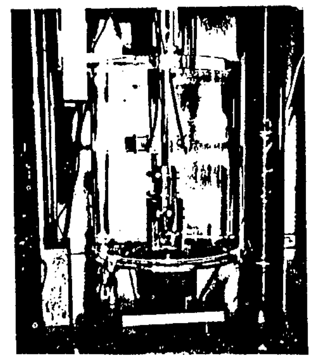
RUSCAN



LOAD FRAME, C-SCAN, AND CONTROLS



C-SCAN/SPECIMEN - FRONT VIEW



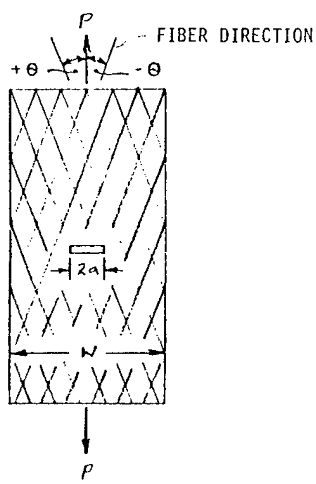
C-SCAN/SPECIMEN - SIDE VIEW

SPECIMEN SCHEMATIC:

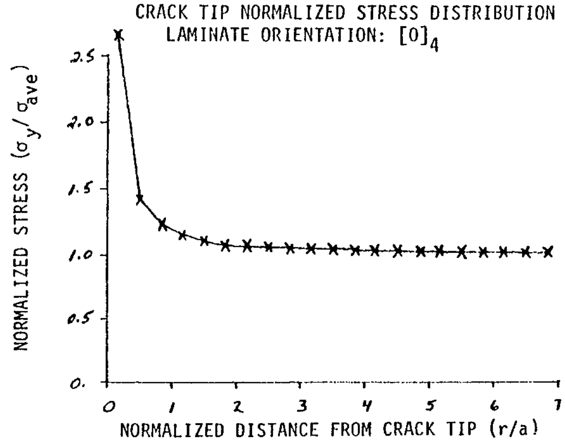
ANGLEPLIED LAMINATE

T300/EPOXY

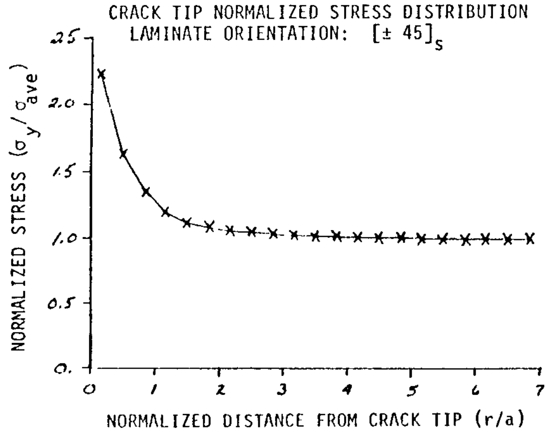
2a = 0.25"
w = 2.00"



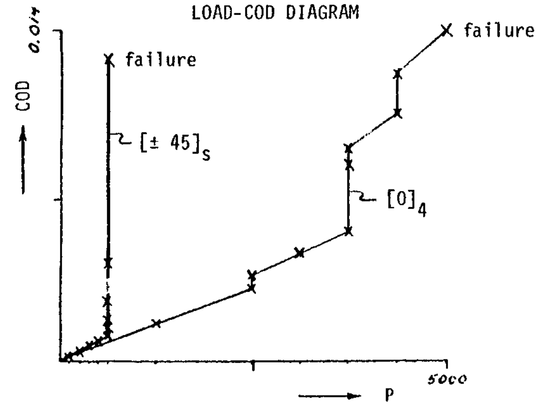
CRACK TIP NORMALIZED STRESS DISTRIBUTION
LAMINATE ORIENTATION: $[0]_4$



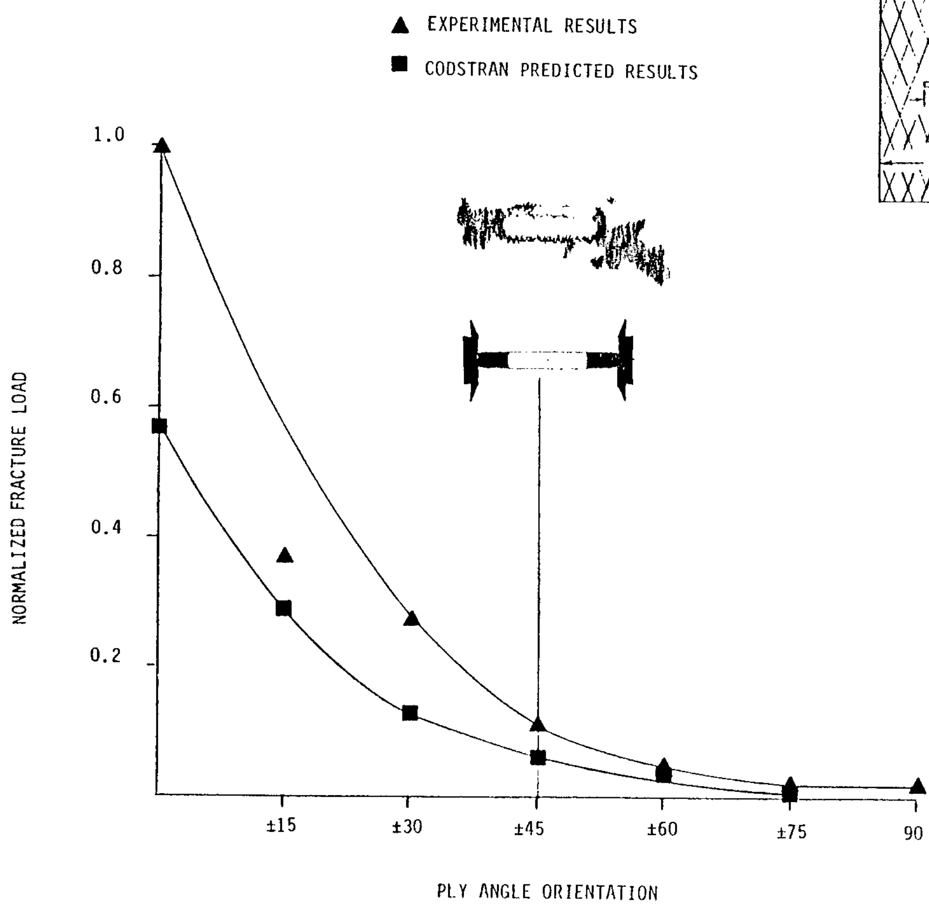
CRACK TIP NORMALIZED STRESS DISTRIBUTION
LAMINATE ORIENTATION: $[\pm 45]_s$

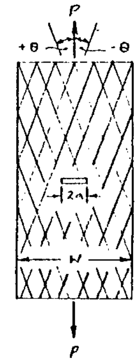


LOAD-COD DIAGRAM



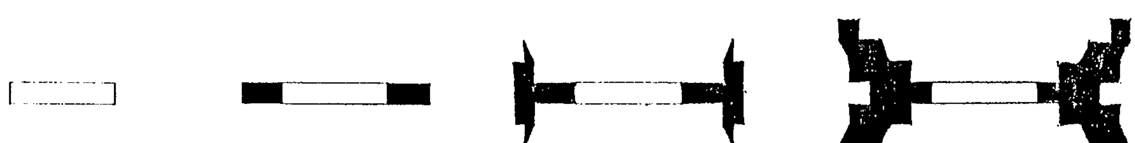
LAMINATE STRENGTH





PROGRESSIVE FRACTURE OF $[\pm 45]_S$ LAMINATE

CODSTRAN Generated Results



NO LOAD

8^{th} ITERATION (80% FRACTURE LOAD)

9^{th} ITERATION (100% FRACTURE LOAD)

10^{th} ITERATION (100% FRACTURE LOAD)

RUSCAN Experimental Results

NO LOAD

80% FRACTURE LOAD

91% FRACTURE LOAD

ANALYSIS OF PROGRESSIVE MATRIX CRACKING IN COMPOSITE LAMINATES

BY

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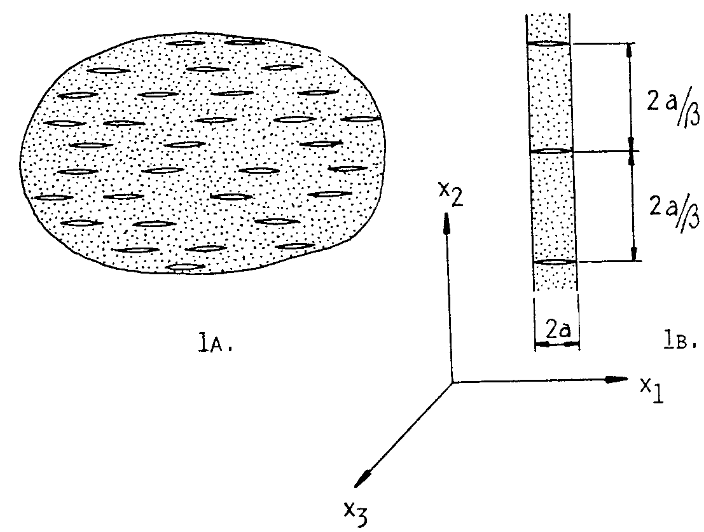
Norman Laws,[2]
Mehdi Hejazi[1]

1 University of Utah, Salt Lake City, UT

2 Cranfield Institute of Technology, Cranfield, England

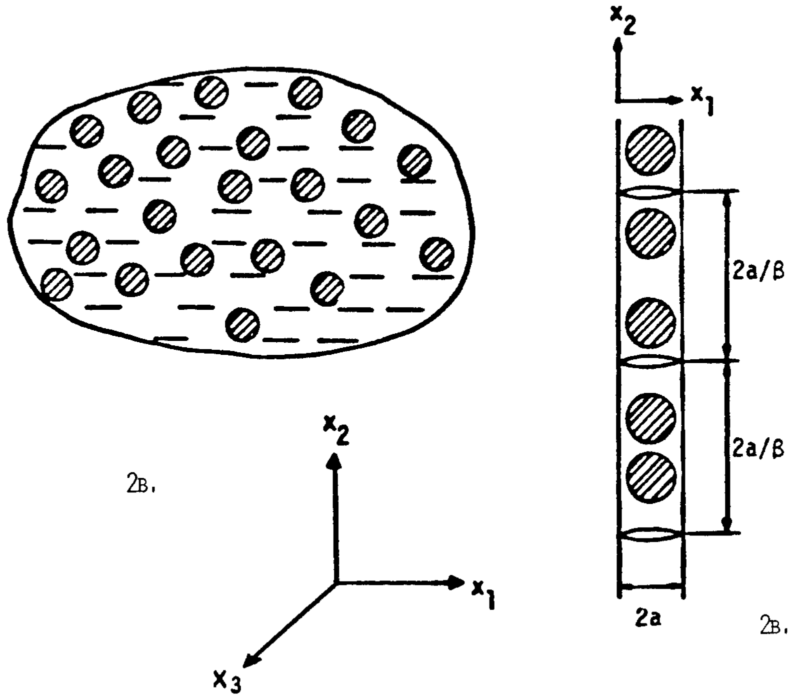
TWO-PHASE MODEL OF A CRACKED LAMINA



1A AN INFINITE FIBROUS MEDIUM WITH ALIGNED SLIT CRACKS.

1B A FIBER LAMINA WITH PARALLEL SLIT CRACKS.

THREE-PHASE MODEL OF A CRACKED LAMINA



2A AN INFINITE MEDIUM WITH ALIGNED FIBERS AND SLIT CRACKS.

2B A FIBER MONOLAYER WITH CRACKS.

TWO AND THREE PHASE SYSTEMS

PHASES	TWO PHASE	THREE PHASE
1	- -	FIBER
2	FIBROUS COMPOSITE "MATRIX"	MATRIX
3	CRACKS	CRACKS

CRACK DENSITY

VOIDS (PHASE 3):

$$\frac{x_1^2}{a^2} + \frac{x_2^2}{b^2} = 1 \quad , \quad |x_3| < 0$$

DENOTE: η — NUMBER OF VOIDS PER UNIT AREA OF x_1x_2-PLANE

$\delta = \beta/a$ — ASPECT RATIO OF VOIDS

$c_3 = \pi ab = \pi a^2 \eta \delta$ — VOLUME FRACTION OF VOIDS

CRACKS (PHASE 3):

DENOTE $c_3 = \frac{1}{4}\pi\beta\delta$ - VOLUME FRACTION OF CRACKS

$\beta = 4\eta a^2$ - CRACK DENSITY PARAMETER

- NUMBER OF CRACKS OF LENGTH $2a$ IN SQUARE OF SIDE $2a$.

$0 \leq \beta \leq 1$: $\beta = 0$ - CRACKS ARE ABSENT

$\beta = 1$ - IN A LAYER OF THICKNESS $2a$, DISTANCE BETWEEN CRACKS IS $2a$.

CRACKS IN A 1/8 MM THICK PLY*

($2a$ = 0.0125 CM)

CRACK DENSITY β	0.1	0.25	0.5	0.75	1.0
DISTANCE BETWEEN CRACKS, $2a/\beta$ (CM)	0.125	0.050	0.025	0.017	0.0125
NUMBER OF CRACKS/UNIT PLANE AREA OF PLY					
# CRACKS/CM^2	8	20	40	60	80
# CRACKS/IN^2	20	50	100	150	203
# CRACKS/FT^2	244	610	1220	1830	2438
NUMBER OF CRACKS/UNIT VOLUME OF PLY					
# CRACKS/CM^3	64	160	320	480	640
# CRACKS/IN^3	412	1030	2060	3090	4120

CRACK DENSITY CHANGE $\Delta\beta$ DUE TO SINGLE CRACK IN UNIT AREA OF PLY

AREA:		
	1 CM^2	$\Delta\beta$ = 0.01250
	1 IN^2	$\Delta\beta$ = 0.00492
	1 FT^2	$\Delta\beta$ = 0.00041

* NOTE THAT β = 0.25-0.28 CORRESPONDS TO SATURATION DENSITY IN 90° DEGREE PLIES IN GR-EP LAMINATES [6]. HOWEVER, CRACK DENSITIES APPROACHING β = 1 WERE OBSERVED IN B-Aℓ PLATES [4].

STIFFNESS CHANGES CAUSED BY CRACKS (1)

GOVERNING EQUATIONS

LINEAR ELASTIC SOLID - OVERALL PROPERTIES

$$\underset{\sim}{\sigma} = L\underset{\sim}{\varepsilon} , \quad \underset{\sim}{\varepsilon} = M\underset{\sim}{\sigma}$$

$$M = L^{-1}$$

ASSUME THAT SOLID CONSISTS OF r PHASES:

$$\sum c_r = 1 \quad , \quad \bar{\underset{\sim}{\sigma}} = \sum c_r \bar{\underset{\sim}{\sigma}}_r \quad , \quad \bar{\underset{\sim}{\varepsilon}} = \sum c_r \bar{\underset{\sim}{\varepsilon}}_r$$

WHERE $\bar{\underset{\sim}{\sigma}}_r$, $\bar{\underset{\sim}{\varepsilon}}_r$ ARE PHASE AVERAGES AND $\sum c_r = 1$.

RELATIONS BETWEEN LOCAL AND OVERALL AVERAGES

$$\bar{\underset{\sim}{\varepsilon}}_r = A\bar{\underset{\sim}{\varepsilon}} \quad , \quad \bar{\underset{\sim}{\sigma}}_r = B_r\bar{\underset{\sim}{\sigma}}$$

A_r - STRAIN CONCENTRATION FACTORS

B_r - STRESS CONCENTRATION FACTORS

FOUND FROM SOLUTION OF AN INCLUSION PROBLEM.

OVERALL STIFFNESS AND COMPLIANCES:

$$L = \sum c_r L_r A_r \quad , \quad M = \sum c_r M_r B_r$$

STIFFNESS CHANGES CAUSED BY CRACKS (2)

A THREE PHASE SYSTEM

$$L = c_1L_1A_1 + c_2L_2A_2 + c_3L_3A_3 \quad , \quad M = c_1M_1B_1 + c_2M_2B_2 + c_3M_3B_3$$

$$c_1A_1 + c_2A_2 + c_3A_3 = I \quad , \quad c_1B_1 + c_2B_2 + c_3B_3 = I$$

Eliminate A_2, B_2:

$$L = L_2+c_1(L_1-L_2)A_1+c_3(L_3-L_2)A_3 , \quad M = M_2+c_1(M_1-M_2)B_1+c_3(M_3-M_2)B_3$$

Introduce A_1, A_3, B_1, B_3 from solution of inclusion problems

$$A_1 = [I + P_1(L_1 - L)]^{-1} \quad , \quad B_1 = [I + Q_1(M_1 - M)]^{-1}$$

$$A_3 = [I + P_3(L_3 - L)]^{-1} \quad , \quad B_3 = [I + Q_3(M_3 - M)]^{-1}$$

Obtain overall L, M of three phase system:

$$L = L_2 + c_1(L_1-L_2)[I+P_1(L_1-L)]^{-1} + c_3(L_3-L_2)[I+P_3(L_3-L)]^{-1}$$

$$M = M_2 + c_1(M_1-M_2)[I+Q_1(M_1-M)]^{-1} + c_3(M_3-M_2)[I+Q_3(M_3-M)]^{-1}$$

With $P_rL + Q_rM = 1$, $(r = 1,3)$

STIFFNESS CHANGES CAUSED BY CRACKS (3)

THREE PHASE SYSTEM WITH VOIDS

Phase 1 - cylindrical fibers

Phase 2 - continuous matrix

Phase 3 - voids, or slit cracks

$$M_3 \to \infty \;,\; L_3 \to 0$$

Express $A_2 B_2$ in terms of A_1, A_3

$$\bar{\underset{\sim}{\varepsilon}}_2 = A_2 \bar{\underset{\sim}{\varepsilon}} = \frac{1}{c_2}(I - c_1 A_1 - c_3 A_3)\bar{\underset{\sim}{\varepsilon}}$$

$$\bar{\underset{\sim}{\sigma}}_2 = L_2 \bar{\underset{\sim}{\varepsilon}}_2 = \frac{1}{c_2} L_2 (I - c_1 A_1 - c_3 A_3) M \bar{\underset{\sim}{\sigma}}$$

With $L_3 \to 0$:

$$A_3 = [I - P_3 L]^{-1} = Q_3^{-1} L$$

Hence

$$A_2 = \frac{1}{c_2}(I - c_1 A_1 - c_3 Q_3^{-1} L)$$

Overall properties of composite with voids:

$$L = L_2 + c_1 (L_1 - L_2)\left[I + P_1 (L_1 - L)\right]^{-1} - c_3 L_2 Q_3^{-1} L$$

$$M = M_2 + c_1 (M_1 - M_2)\left[I + Q_1 (M_1 - M)\right]^{-1} + c_3 Q_3^{-1}$$

STIFFNESS CHANGES CAUSED BY CRACKS (4)

THREE PHASE SYSTEM WITH CRACKS

Evaluation of L, M, for $\delta \to 0$

Need to find

$$\lim_{\delta \to 0} \delta Q_3^{-1} = \Lambda$$

Then

$$\boxed{\begin{aligned} L &= L_2 + c_1 (L_1 - L_2)\left[I + P_1 (L_1 - L)\right]^{-1} - \tfrac{1}{4}\pi \beta L_2 \Lambda \\ M &= M_2 + c_1 (M_1 - M_2)\left[I + Q_1 (M_1 - M)\right]^{-1} + \tfrac{1}{4}\pi \beta \Lambda \end{aligned}}$$

Where (Laws 1977)

$$\Lambda_{1212} = \frac{(L_{11} L_{22})^{\frac{1}{2}} (\alpha_1^{\frac{1}{2}} + \alpha_2^{\frac{1}{2}})}{4(L_{11} L_{22} - L_{12}^2)}$$

$$\Lambda_{2222} = \frac{L_{11} (\alpha_1^{\frac{1}{2}} + \alpha_2^{\frac{1}{2}})}{L_{11} L_{22} - L_{12}^2}$$

$$\Lambda_{2323} = \frac{1}{4(L_{44} L_{55})^{\frac{1}{2}}}$$

α_1, α_2 are roots of

$$L_{11} L_{66} \alpha^2 - (L_{11} L_{22} - L_{12}^2 - 2 L_{12} L_{66})\alpha + L_{22} L_{66} = 0$$

TWO PHASE SYSTEM WITH CRACKS

Phase 1 - not present

Phase 2 - fibrous composite is now "matrix"

Phase 3 - cracks

Overall Properties

$$\boxed{\begin{aligned} L &= L_2 - \tfrac{1}{4}\pi\beta L_2 \Lambda L \\ M &= M_2 + \tfrac{1}{4}\pi\beta\Lambda \end{aligned}}$$

Tensor Λ in terms of compliances M_{ij}

$$\Lambda_{22} = \Lambda_{2222} = \frac{M_{22}M_{33}-M_{23}^2}{M_{33}}\left(\alpha_1^{\frac{1}{2}} + \alpha_2^{\frac{1}{2}}\right) \qquad \Lambda_{44} = 4\Lambda_{2323} = \left(M_{44}M_{55}\right)^{\frac{1}{2}}$$

$$\Lambda_{66} = 4\Lambda_{1212} = \frac{\left(M_{22}M_{33}-M_{23}^2\right)^{\frac{1}{2}}\left(M_{11}M_{33}-M_{13}^2\right)^{\frac{1}{2}}}{M_{33}}\left(\alpha_1^{\frac{1}{2}} + \alpha_2^{\frac{1}{2}}\right)$$

Unchanged compliances:

$$M_{11} = M_{11}^{(2)} \ , \quad M_{33} = M_{33}^{(2)} \ , \quad M_{55} = M_{55}^{(2)}$$

$$M_{12} = M_{12}^{(2)} \ , \quad M_{13} = M_{13}^{(2)} \ , \quad M_{23} = M_{23}^{(2)}$$

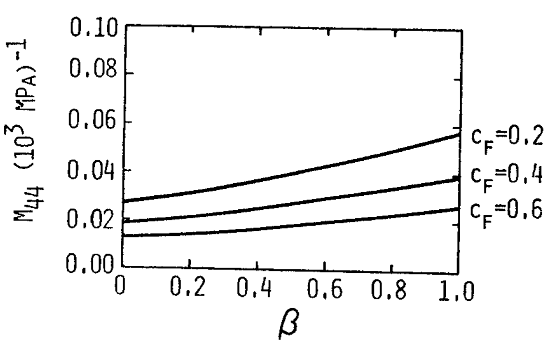
Changed compliances:

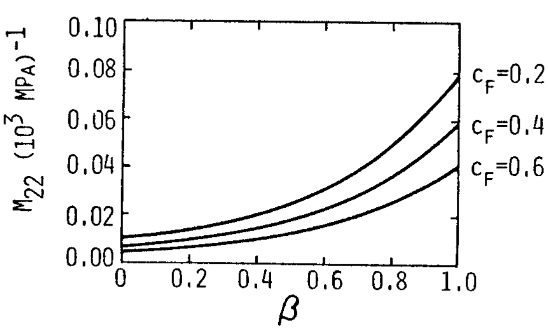
$$M_{22} = M_{22}^{(2)} + \tfrac{1}{4}\pi\beta\Lambda_{22}$$

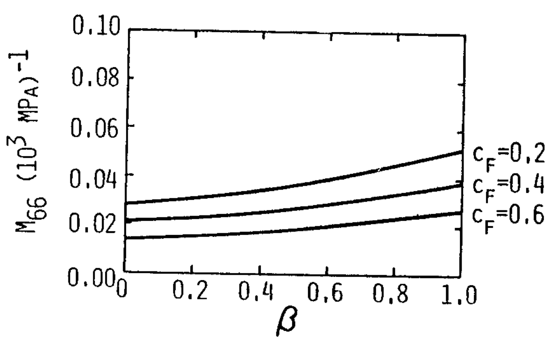
$$M_{44} = M_{44}^{(2)} + \tfrac{1}{4}\pi\beta\Lambda_{44}$$

$$M_{66} = M_{66}^{(2)} + \tfrac{1}{4}\pi\beta\Lambda_{66}$$

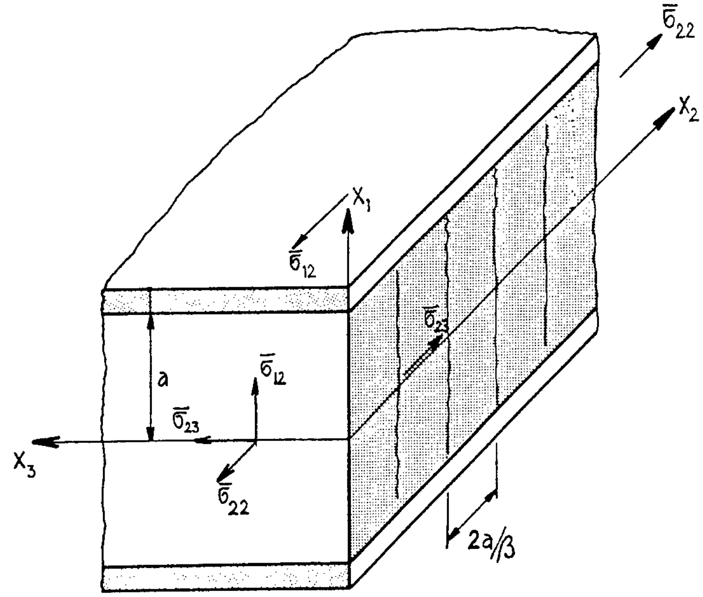
COMPLIANCE CHANGES IN A GR/EP SYSTEM







CRACK GROWTH IN FIBROUS LAMINA (1)



GEOMETRY OF MULTIPLE TRANSVERSE CRACKS.

CRACK GROWTH IN FIBROUS LAMINA (2)

Extension of initial flaws in x_1x_3-plane:

Critical stresses for lamina of thickness 2a, at current crack density

Mode I: $\bar{\sigma}_{22} = \bar{\sigma}_I(a,\beta)$

Mode II: $\bar{\sigma}_{12} = \bar{\sigma}_{II}(a,\beta)$

Mode III: $\bar{\sigma}_{23} = \bar{\sigma}_{III}(a,\beta)$

Failure criterion

$$\left(\frac{\bar{\sigma}_{22}}{\bar{\sigma}_I}\right)^2 + \left(\frac{\bar{\sigma}_{12}}{\bar{\sigma}_{II}}\right)^2 + \left(\frac{\bar{\sigma}_{23}}{\bar{\sigma}_{III}}\right)^2 = 1$$

$\bar{\sigma}_I$, $\bar{\sigma}_{II}$, $\bar{\sigma}_{III}$ are found from experiment or independent analysis. They are assumed to be monotonically increasing functions of β.

CRACK GROWTH IN FIBROUS LAMINA (3)

Strain softening

Increase in crack density with applied overall strain $d\bar{\underset{\sim}{\varepsilon}}$

New Strain: $\bar{\underset{\sim}{\varepsilon}}' = \bar{\underset{\sim}{\varepsilon}} + d\bar{\underset{\sim}{\varepsilon}}$

New Stress: $\bar{\underset{\sim}{\sigma}}' = \underset{\sim}{L}' \, \bar{\underset{\sim}{\varepsilon}}'$

Failure Criterion:

$$\left(\frac{\bar{\sigma}_{22}}{\bar{\sigma}_{I}}\right)^2 + \left(\frac{\bar{\sigma}_{12}}{\bar{\sigma}_{II}}\right)^2 + \left(\frac{\bar{\sigma}_{23}}{\bar{\sigma}_{III}}\right)^2 = 1$$

New stress components must satisfy failure criterion

$$\bar{\sigma}'_{12} = 2L'_{66}\,\bar{\varepsilon}'_{12}$$

$$\bar{\sigma}'_{22} = L'_{12}\,\bar{\varepsilon}'_{11} + L'_{22}\,\bar{\varepsilon}'_{22} + L'_{23}\,\bar{\varepsilon}'_{33}$$

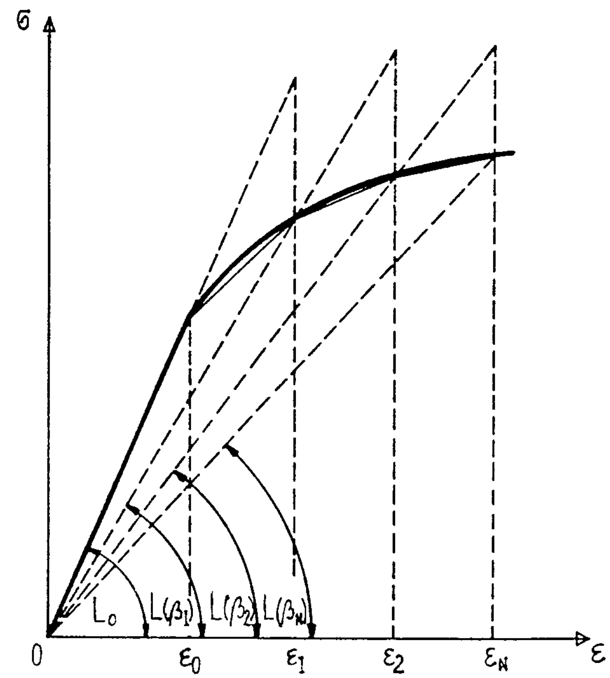
$$\bar{\sigma}'_{23} = 2L'_{44}\,\bar{\varepsilon}'_{23}$$

Hence:

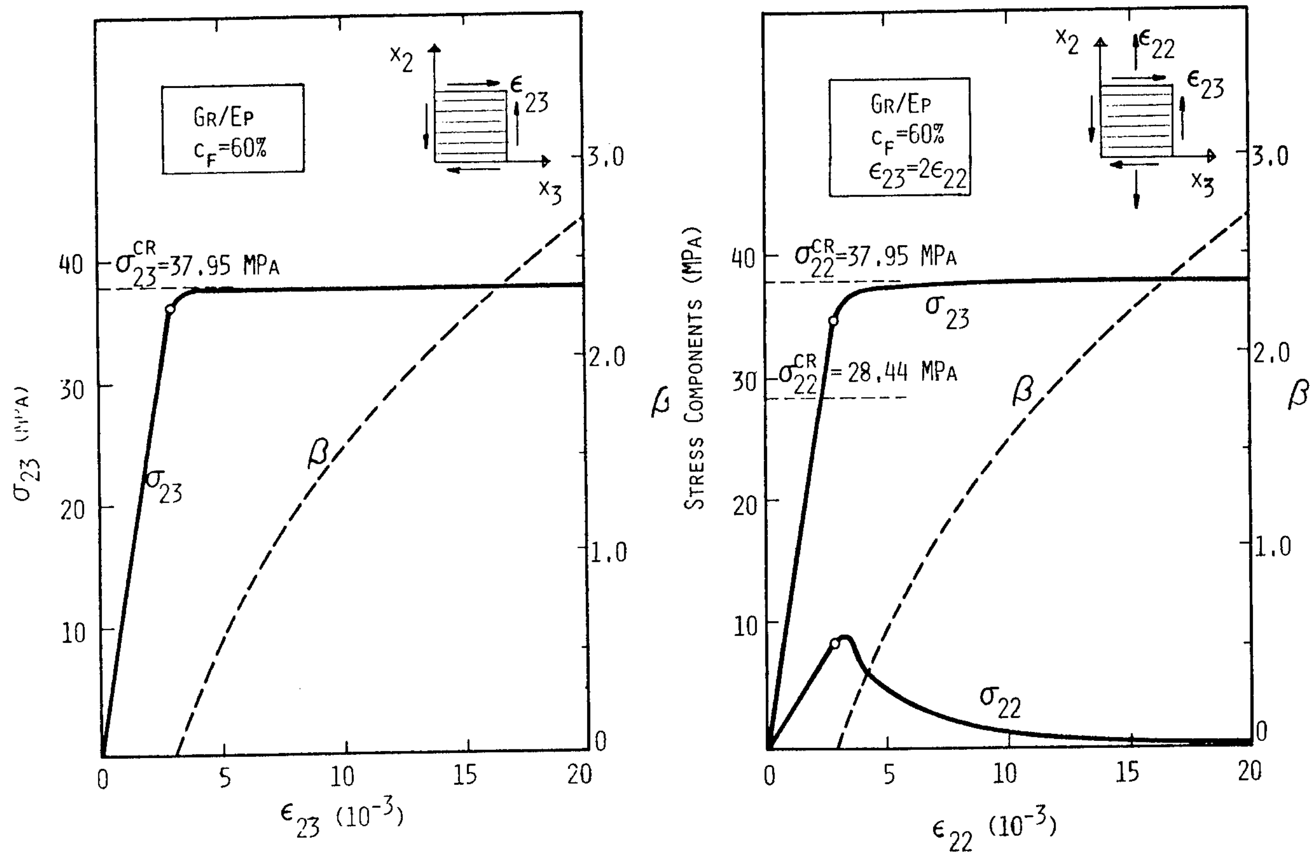
$$\left(\frac{L'_{12}\,\bar{\varepsilon}'_{11} + L'_{22}\,\bar{\varepsilon}'_{22} + L'_{23}\,\bar{\varepsilon}'_{33}}{\bar{\sigma}_{I}}\right)^2 + \left(\frac{2L'_{66}\,\varepsilon'_{12}}{\bar{\sigma}_{II}}\right)^2 + \left(\frac{2L'_{44}\,\varepsilon'_{23}}{\bar{\sigma}_{III}}\right)^2 = 1$$

An increment $d\beta$ must be found such that L'_{ij} satisfy the failure criterion.

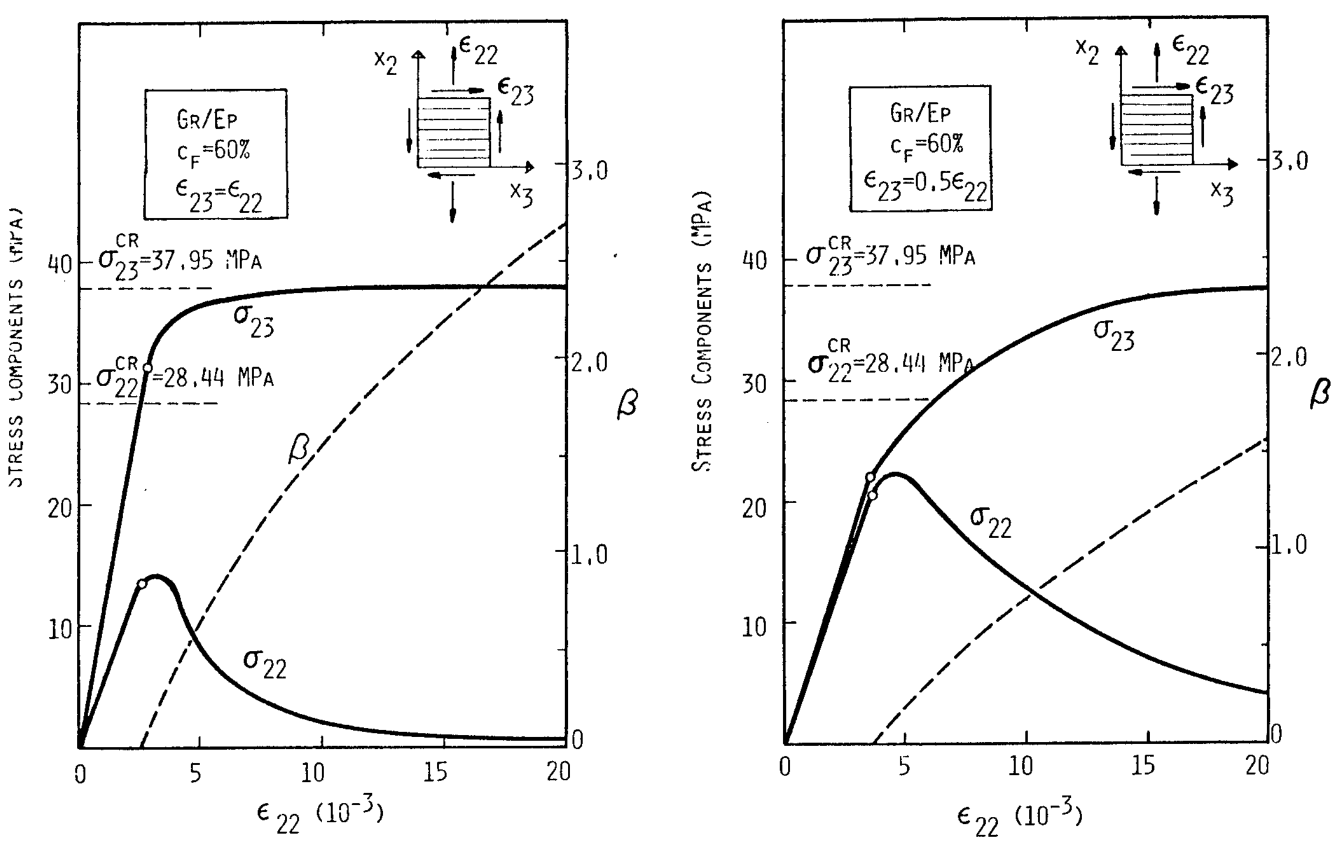
STIFFNESS CHANGE IN CRACKING LAMINA



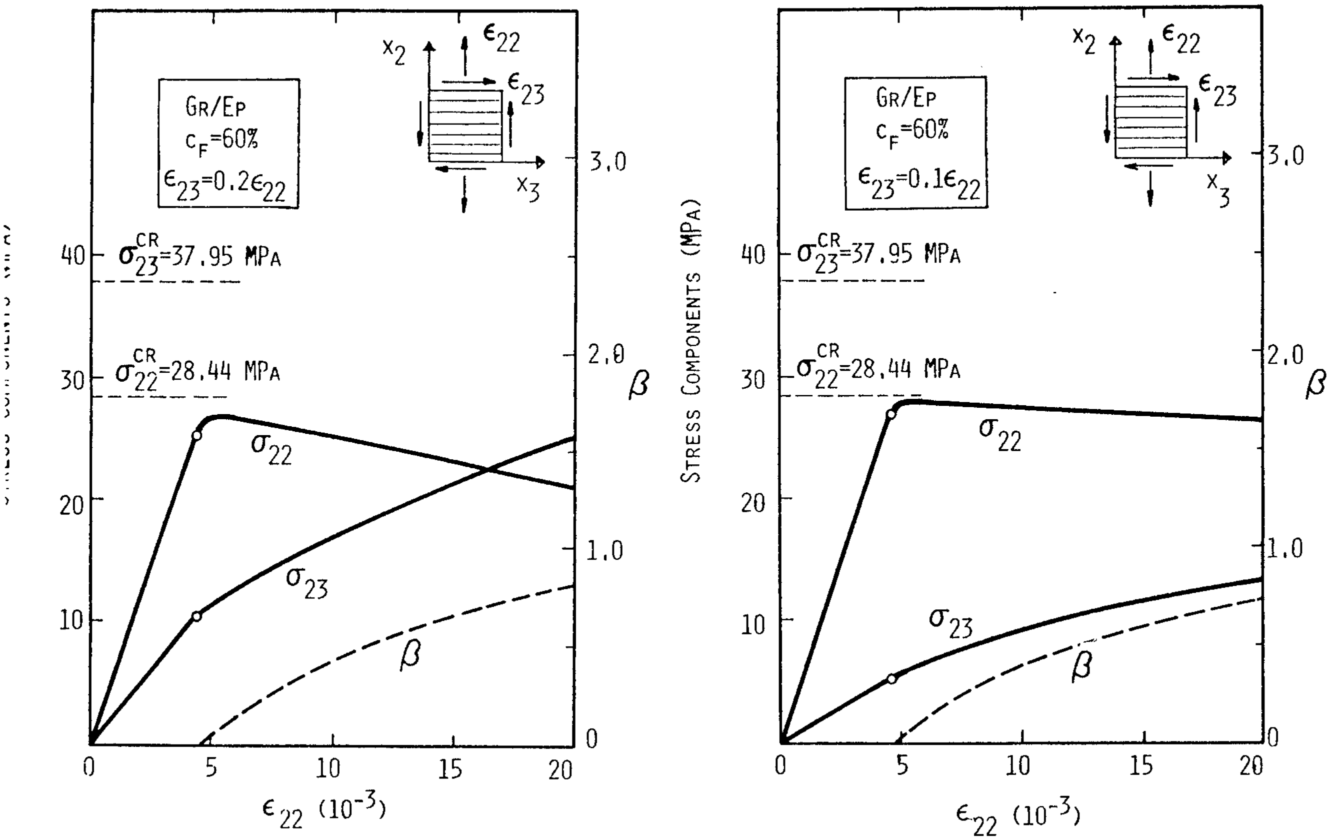
SCHEMATIC REPRESENTATION OF EVALUATION OF STIFFNESS CHANGES WITH INCREASING CRACK DENSITY β IN A FIBROUS LAMINA.



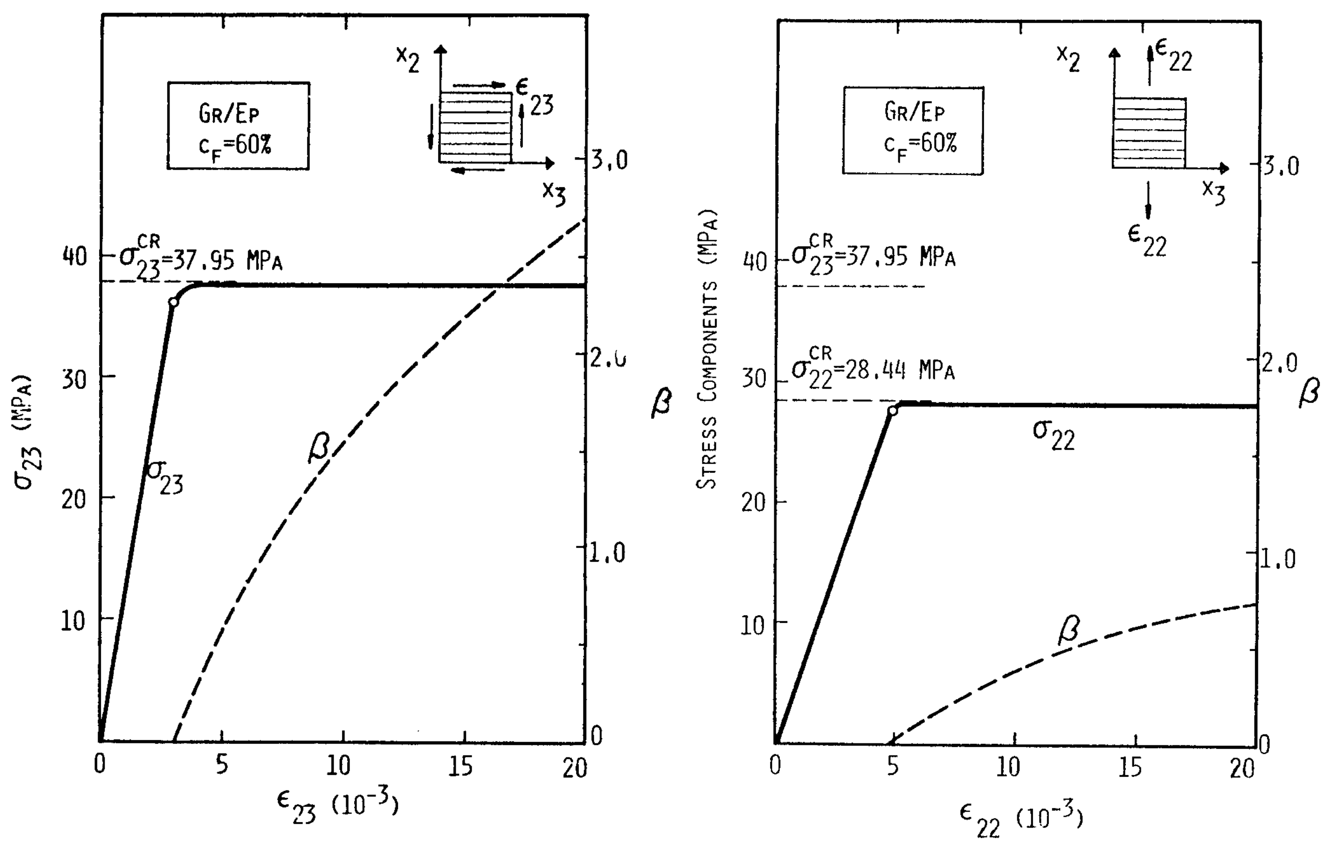
STRESSES IN A CRACKING LAMINA DURING PRESCRIBED PROPORTIONAL STRAINING. PLANE STRESS, $\sigma_{11} = \sigma_{33} = 0$.



STRESSES IN A CRACKING LAMINA DURING PRESCRIBED PROPORTIONAL STRAINING. PLANE STRESS, $\sigma_{11} = \sigma_{33} = 0$.

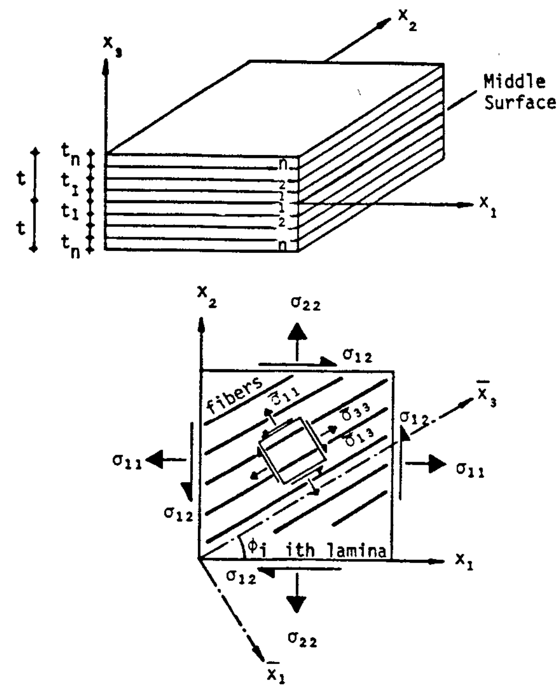


STRESSES IN A CRACKING LAMINA DURING PRESCRIBED PROPORTIONAL STRAINING, PLANE STRESS, $\sigma_{11} = \sigma_{33} = 0$.



STRESSES IN A CRACKING LAMINA DURING PRESCRIBED PROPORTIONAL STRAINING, PLANE STRESS, $\sigma_{11} = \sigma_{33} = 0$.

CRACK GROWTH IN LAMINATED PLATES (1)



A BALANCED, SYMMETRIC LAMINATED PLATE UNDER IN-PLANE STRESSES.

1. <u>LOCAL AND OVERALL COORDINATES</u>

LOCAL LAMINA COORDINATES x_i.

OVERALL LAMINA AND LAMINATE COORDINATES x_j.

CRACK GROWTH IN LAMINATED PLATES (2) - CONTINUED

2. <u>Local and overall constitutive equations</u>:

<u>Lamina</u>:

Local $\quad \bar{\sigma}_i = \bar{L}_i \bar{\varepsilon}_i \,, \quad \bar{\varepsilon}_i = \bar{M}_i \bar{\sigma}_i$

Overall $\quad \bar{\sigma}_i = T_i \sigma_i \,, \quad \bar{\varepsilon}_i = N_i \varepsilon_i$

$\sigma_i = L_i \varepsilon_i \,, \quad \varepsilon_i = M_i \sigma_i$

<u>Laminate</u>: $\quad \sigma = L\varepsilon \qquad \varepsilon = M\sigma$

<u>Compatibility and equilibrium</u>

$$\varepsilon = \varepsilon_1 = \varepsilon_2 = \dots \varepsilon_n$$

$$\sigma = c_1\sigma_1 + c_2\sigma_2 + \dots c_n\sigma_n \,, \qquad c_i = t_i/t$$

<u>Stress distribution factors</u>

$$\sigma_i = H_i \sigma \,, \qquad c_1 H_1 + c_2 H + \dots c_n H_n = I$$

$$H_i = L_i L^{-1} = M_i^{-1} M$$

<u>Overall stiffness L and compliance M</u>

$$L = c_1 L_1 + c_2 L_2 + \dots c_n L_n$$

$$M = M_1 H_1 = M_2 H_2 = \dots M_n H_n$$

3. Fracture criterion in each lamina (local coordinates)

$$F_i(\bar{\sigma}_i) = 0 \rightarrow \text{Obtain new } \beta$$

$$\bar{\sigma}_i = \bar{L}_i(\beta)\, \bar{\varepsilon}_i \,, \quad \bar{\varepsilon}_i = \bar{M}_i(\beta)\, \bar{\sigma}_i$$

CONCLUSIONS

1. Stiffness changes caused in a lamina by distributed cracks of a certain density β can be evaluated from a simple numerical procedure in composites with large diameter fibers, and in closed form in composites with small diameter fibers.

2. Fracture criteria for transverse cracking in a lamina can be utilized, in conjunction with the stiffness change evaluation, in an incremental procedure which gives estimates of average stresses and strains, and crack densities in a ply at each point of a prescribed loading path.

3. The results obtained in 1. and 2., above, can be incorporated into analysis of laminated plates. Crack densities, average stresses and strains in the layers, fiber and matrix stresses in each ply, and macroscopic stiffness changes of the plate can be found.

DAMAGE ACCUMULATION IN COMPOSITES

PERFORMED BY

GENERAL DYNAMICS
FORT WORTH

D. A. ULMAN
R. D. BRUNER
H. R. MILLER

SPONSORED BY

FLIGHT DYNAMICS
LABORATORY

F33615-81-C-3226
G. P. SENDECKYJ
AFWAL/FIBEC

OBJECTIVES

- To Document the Damage Accumulation Process in Composites
- To Measure the Degradation of Laminate Stiffness
- To Develop Life Prediction Procedures for Composite Laminates

APPROACH

TASK I : PRELIMINARY SPECIMEN SCREENING

- TEST FOUR CANDIDATE LAMINATE CONFIGURATIONS
- EVALUATE SPECIMEN DESIGNS AND TEST PROCEDURES

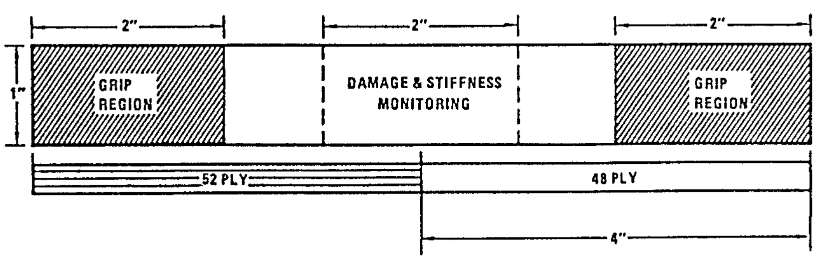
TASK II: LIFE PREDICTION PROCEDURE DEVELOPMENT

- EXTENSIVELY TEST A SINGLE LAMINATE CONFIGURATION
- DEVELOP LIFE PREDICTION PROCEDURES

CONCLUSIONS TO DATE

- o Damage Accumulation is Systematic and Progressive
- o Stiffness Change is An Indicator of Damage
- o Delamination is the Dominant Damage Mode
- o Stress Level Affects Damage Rates
- o Stress Ratio Affects the Damage Accumulation Processes
- o Stress Range Contributes to Damage Growth
- o Tensile Stress Creates Transverse Cracks
- o Compressive Stress in the Presence of Delamination Can Create Local Instabilities Which Lead to Buckling-Type Failures

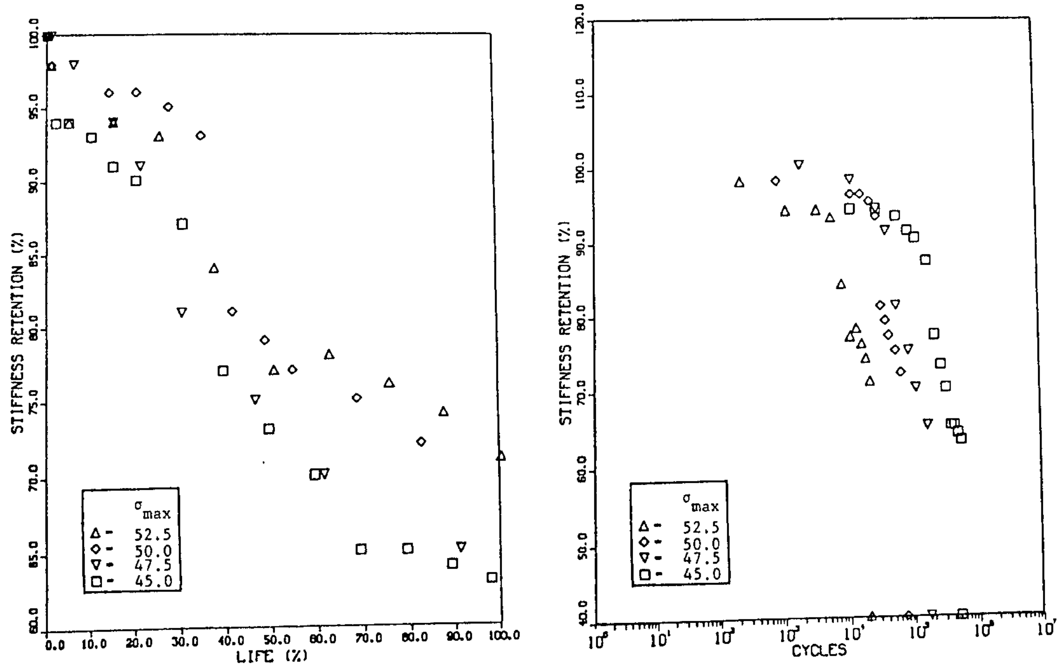
PLY-TERMINATION SPECIMEN



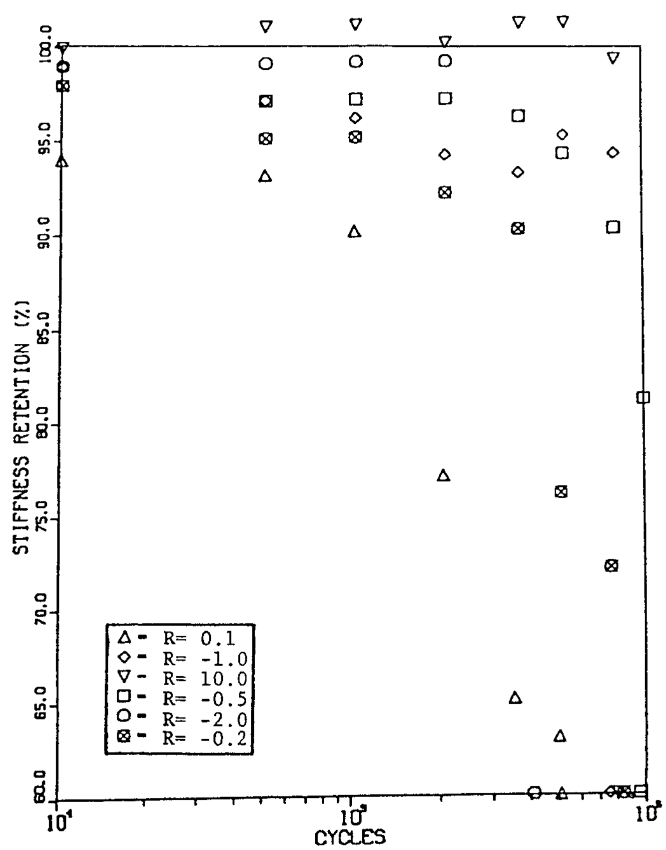
Type	Family	Stacking Sequence*
C	(25/50/25)	$[(0/+/90/-)_S\ (0/-/90/+)_S\ (0/+/90/-)_S]_S$
CT	(25/50/25)	$[(0/+/90/-)_S\ (0/-/90/+/\underline{0})_S\ (0/+/90/-)_S]_S$ └TERMINATED PLIES

* + INDICATES A +45° PLY
– INDICATES A –45° PLY

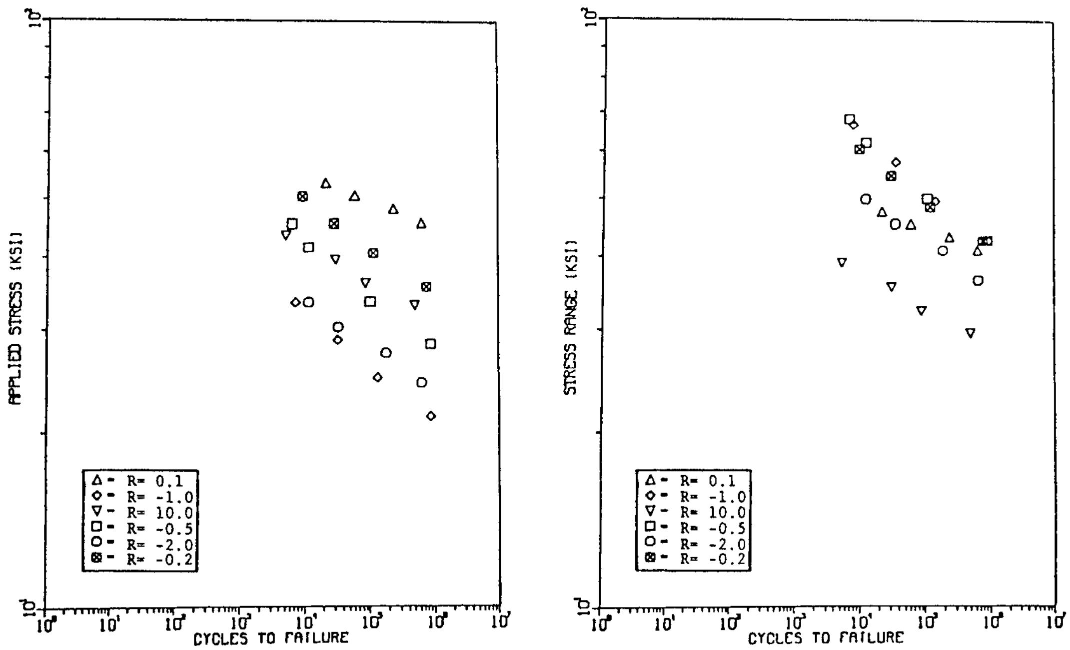
STIFFNESS RETENTION



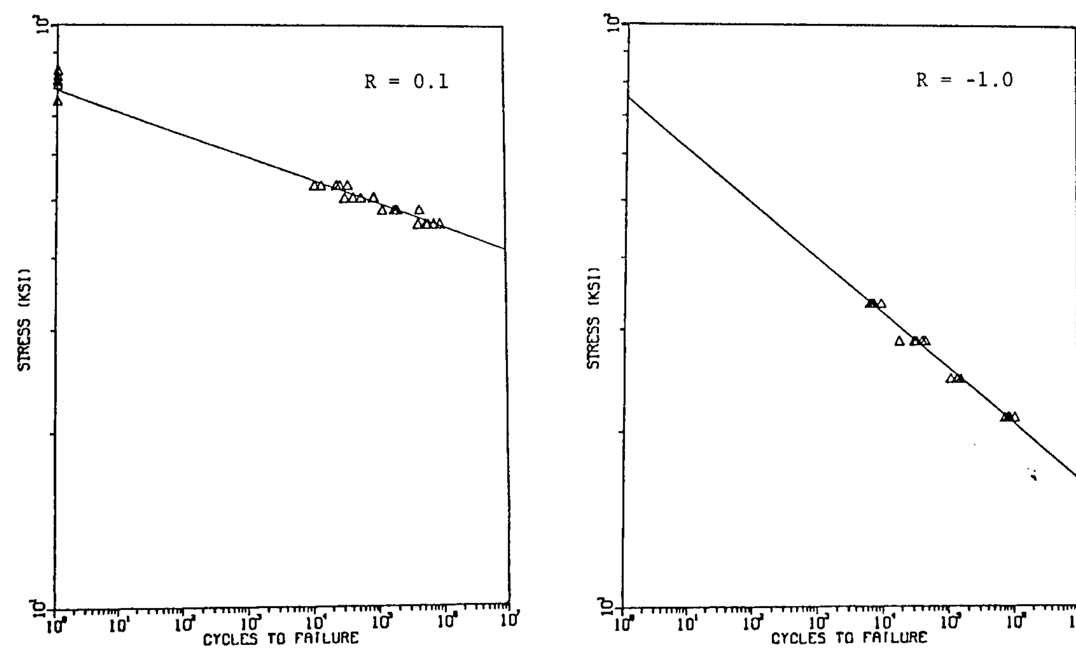
STRESS RATIO EFFECTS ON
STIFFNESS DEGRADATION



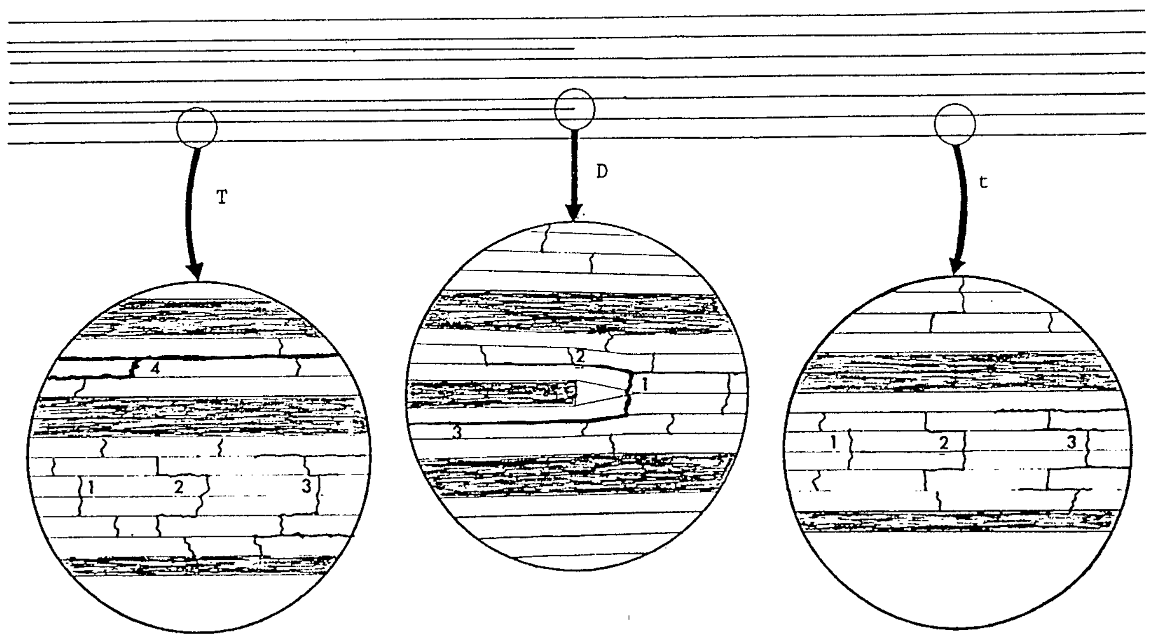
AVERAGE LIFE VERSUS STRESS PARAMETERS



STRESS RATIO EFFECT ON LIFE

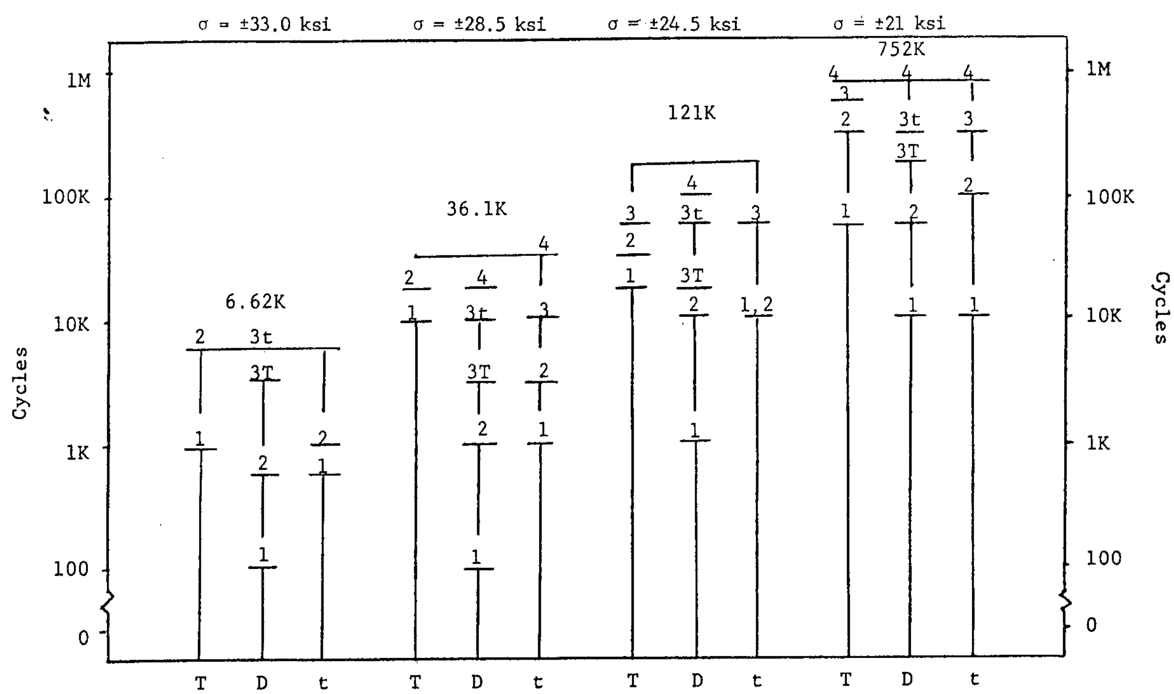


SCHEMATIC OF DAMAGE MODES

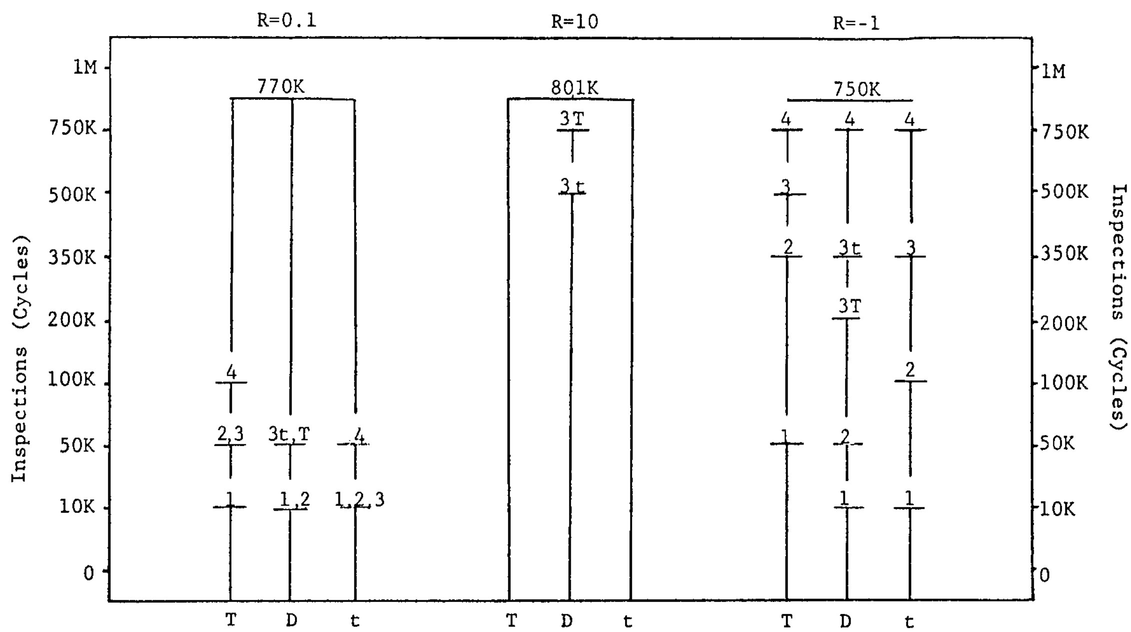


1. PLY CRACKING
2. CRACK COUPLING
3. DELAMINATION
4. SEVERE DELAMINATION

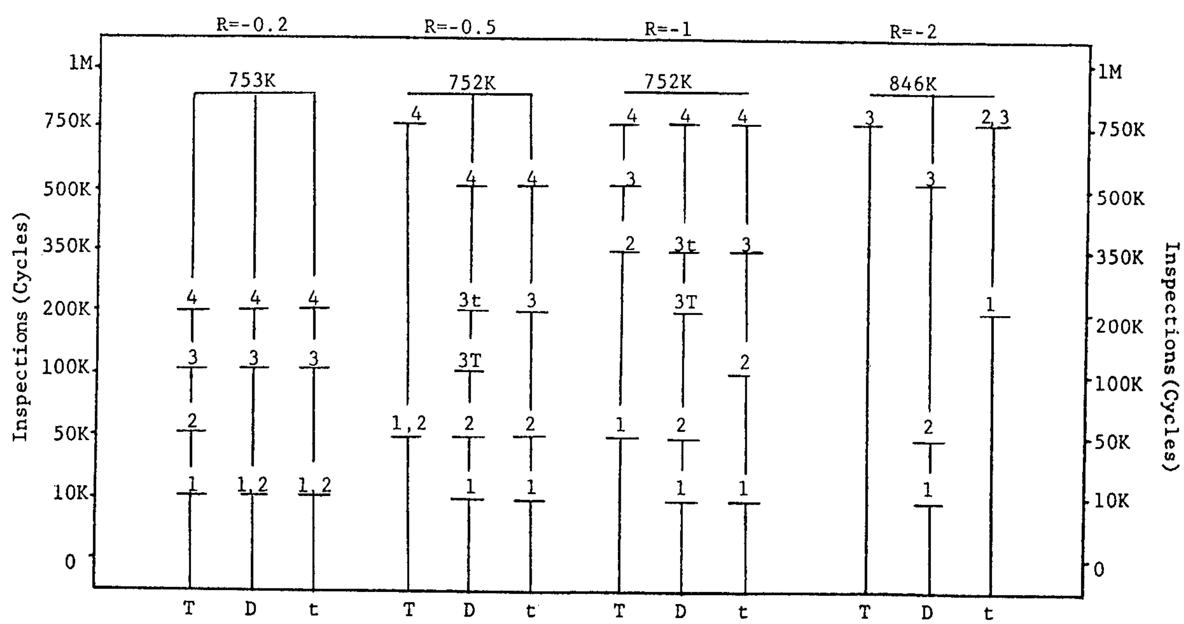
STRESS LEVEL AFFECTS DAMAGE RATE



DAMAGE ACCUMULATION PROCESSES DIFFER



DAMAGE RATES DIFFER



INTERLAMINAR AND INTRALAMINAR FRACTURE GROWTH IN COMPOSITE LAMINATES

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DREXEL UNIVERSITY

* PROJECT SUPPORTED BY AFOSR .

OBJECTIVES

- To study the physical mechanisms of TRANSVERSE CRACKING(intralaminar) and FREE EDGE DELAMINATION(interlaminar) growth processes in graphite-epoxy laminates;
- To develop a general method from which a predictive model is derived for these two types of damages and their interacting effects;
- To conduct experimental case studies in order to correlate both the basic methodology and the predictive model(s).

LAMINATES STUDIED:

o Uniaxial tension:
  - $(\pm 25/90_n)_s$ — $n = 1/2, 1, 2, 3, 4, 6, 8$
  - $(\pm\theta/90_n)_s$ — $n = 1, 2, 4$; $\theta = 30^0, 45^0, 60^0$
  - $(0/90_n/0)_s$ — $n = 1, 2, 3, 4,$
  - $(0_2/90_n/0_2)_s$ — $n = 2, 4, 8.$
  - $(\pm 25_2/90_2)_s$ — $(+45_n)_s$ — $n = 2,3$

o Uniaxial compression:
  - $(0_2/90_2/\pm 45_2)_s$
  - $(90_2/0_2/\pm 45_2)_s$
  - $(0/90/0/90/\pm 45/\pm 45)_s$

CONCLUSIONS

I ON THE PHYSICAL MECHANISMS:

- TRANSVERSE CRACKS and DELAMINATION are MATRIX CRACKS which involve no fiber breakage;
- The formation of a MATRIX CRACKstems from sudden coalesence of inherent material flaws(voids, fiber/matrix disbonds, etc.);
- The orientation of a MATRIX CRACK generally follow the fiber/matrix interface and/or ply to ply interface;
- The propagation of a MATRIX CRACK is a brittle fracture event; and the propagation direction is self-similar;
- Laminate interface can arrest or blunt a propagating MATRIX CRACK hence localizes the damage;
- Multiple MATRIX CRACKS can form in the course of ascending load; a certain characteristic damage pattern is resulted;
- The development of characteristic MATRIX CRACKING pattern under load is a part of laminate DAMAGE ACCUMULATION PROCESS.

II ON THE MODELING METHODOLOGY:

o Ply-Elasticity is the basis for laminate stress analysis;

o The concept of constant ply strength is replaced by the concept of EFFECTIVE FLAW DISTRIBUTION in order to describe multiple MATRIX CRACKS in the laminate;

o MATRIX CRACKS are results of EFFECTIVE FLAW propagations;

CONCLUSIONS-cont'd

o EFFECTIVE FLAWS can be represented by a SIZE DISTRIBUTION and a SPACING DISTRIBUTION; both are material related properties;

o For each EFFECTIVE FLAW to become a MATRIX CRACK the engineering fracture mechanics criterion is applicable;

o The arrest mechanisms of a MATRIX CRACK can be modeled by a crack-growth simulation which takes into account the lamination geometry of the laminate;

o Multiple MATRIX CRACKS under ascending load can be modeled by a STOCHASTIC simulation process which takes into account the EFFECTIVE FLAW distributions in the laminate;

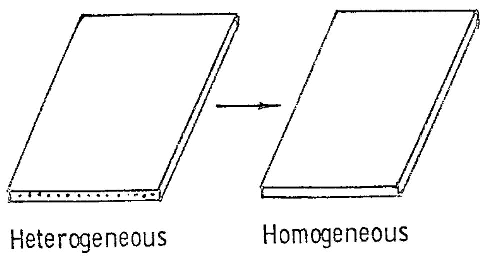
o The fracture mechanics simulation of a propagating MATRIX CRACK can be carried out by a 2-D and/or 3-D finite element routine;

o The stochastic simulation of multiple MATRIX CRACKS can be carried out by a numerical MONTE CARLO random search routine.

THE FRACTURE MECHANICS/STOCHASTIC SIMULATION MODEL HAS CORRELATED WELL THE FOLLOWING MATRIX CRACK GROWTH CASES:

- Multiple transverse cracks in 90-layer; eg. $(0/90)_s$ laminates;
- Free edge delamination in a laminate coupon; eg. $(\pm 25/90)_s$;
- Transverse crack/free edge delamination interaction; eg. $(0_2/90_4)_s$.

CONCEPT OF PLY-ELASTICITY:



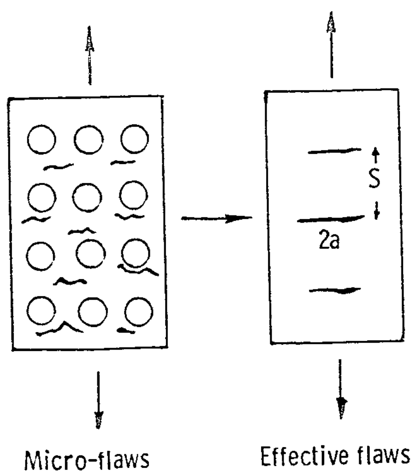
Constitutive:

$\sigma_i = C_{ij} e_j$

Failure:

$F_{ij}\sigma_i\sigma_j + F_i\sigma_i = 1$

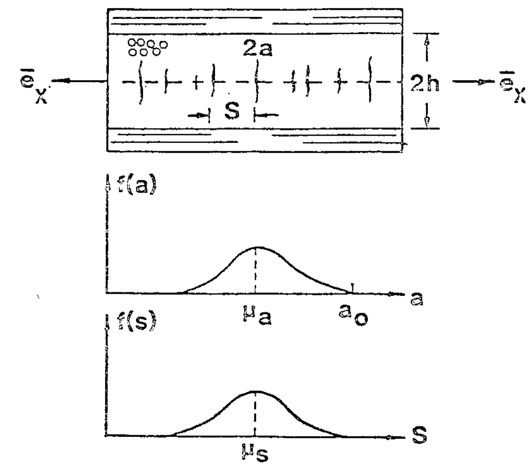
CONCEPT OF EFFECTIVE FLAW DISTRIBUTION:



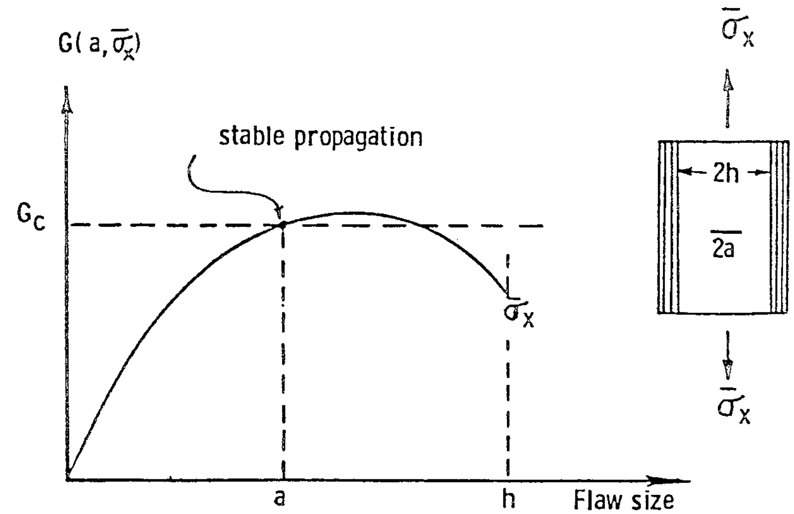
Flaw size 2a and flaw spacing S are random;
Within a representative volume f(a) and f(S) are EFFECTIVE ply properties.

THE FRACTURE MECHANICS /STOCHASTIC MODEL FOR TRANSVERSE CRACKS

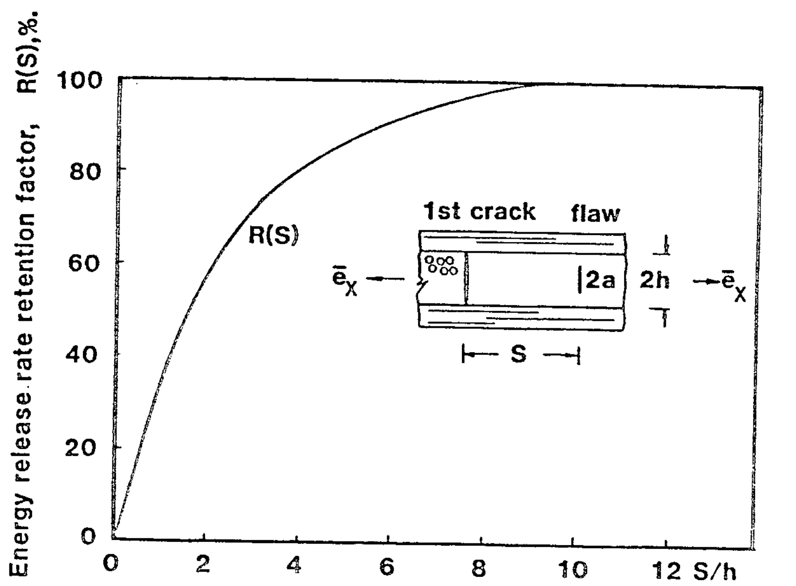
First .....
the effective flaw distributions are assumed known;



Second... .. fracture mechanics criterion is applied to define the condition for a flaw to become a crack. ...



But for multiple cracks we must include the so-called SHEAR-LAG effect. Whin the shear-lag zone of a crack, the condition for a flaw to become a crack is changed(reduced)..



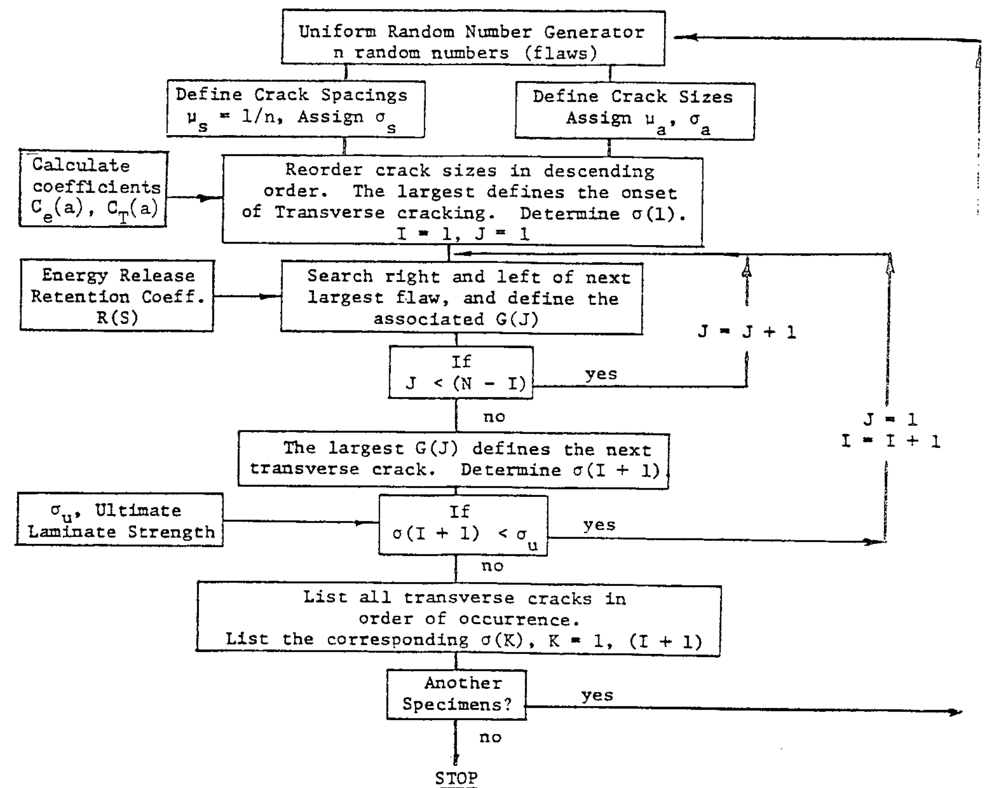
The Retention Factor R(S) is computed by a finite element shear-lag analysis.

Thus, for a flaw located between two cracks, its energy release rate is

$$G(a,\bar{\sigma}_x) = R(S_L)\,\bar{G}(a,\bar{\sigma}_x)\,R(S_R)$$

$\bar{G}(a,\bar{\sigma}_x)$ is the energy release rate W/O the presence of any crack

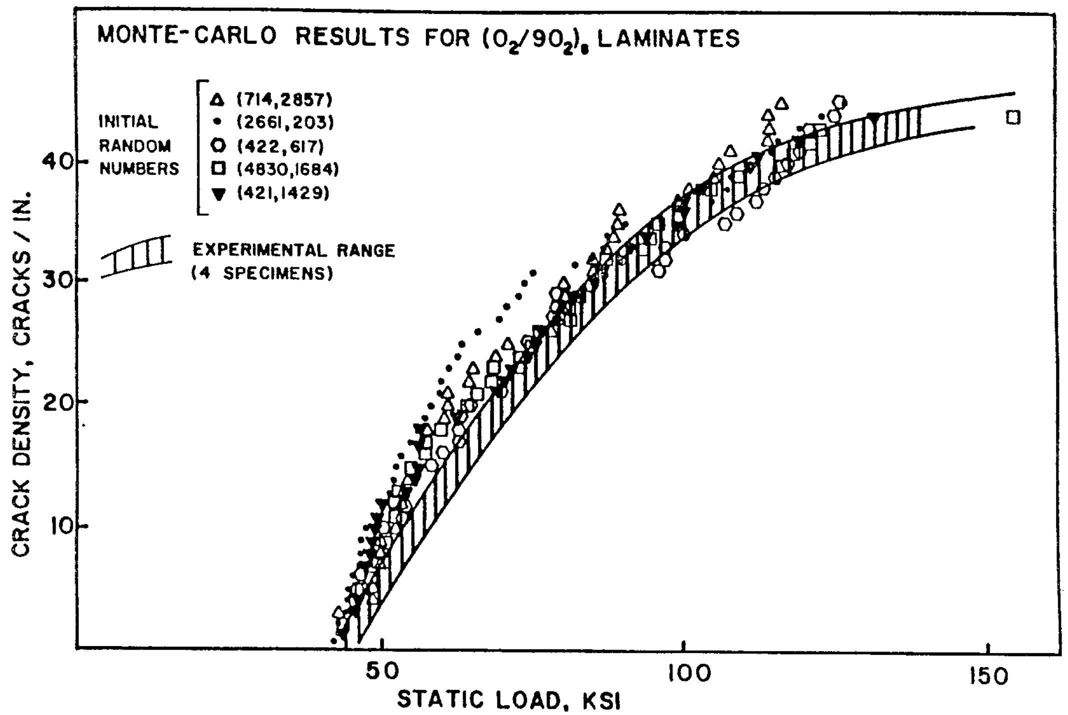
TABLE A FLOW-CHART FOR MONTE-CARLO SIMULATION



EXAMPLE: T300/934 $(0_2/90_2)_s$ laminate under simple tension

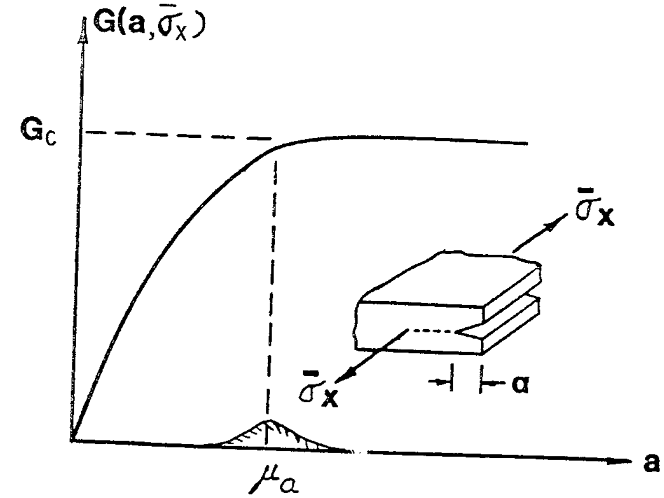
f(a) and f(S) are assumed NORMAL distributions:

$\mu_a = 0.0036''$ $\sigma_a = 0.0013''$ $\mu_s = 0.0125''$ $\sigma_s = 0.0046''$

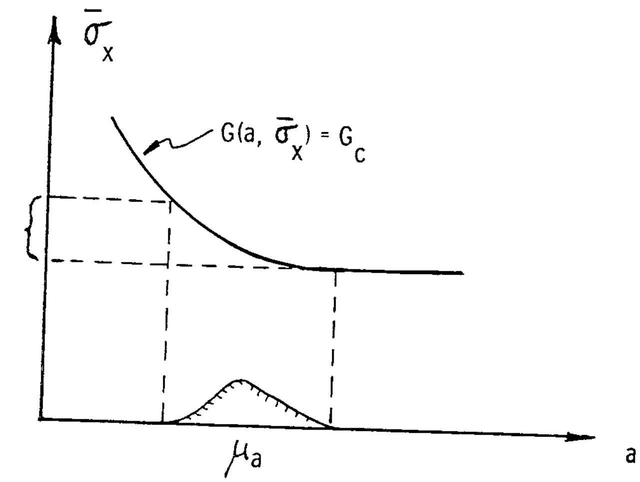


PREDICTIVE MODEL FOR FREE EDGE DELAMINATION

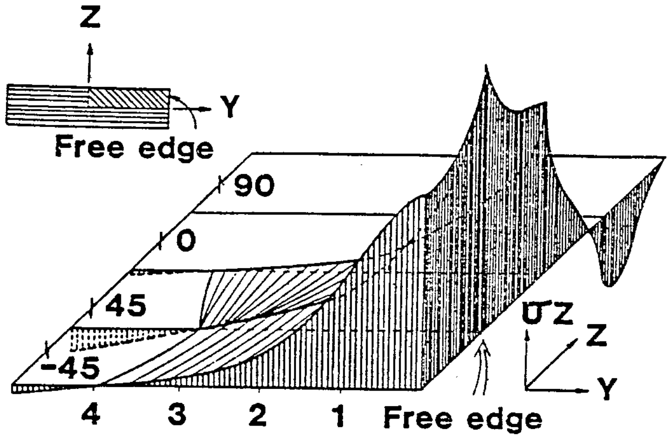
Again, the same ENERGY approach is applied. . . .



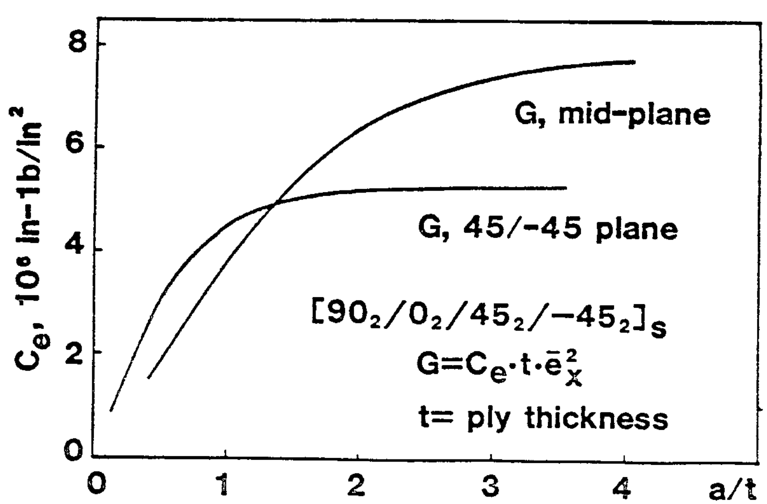
Predicted load range for delamination



EXAMPLE: $(90_2/0_2/45_2/-45_2)_s$ under simple compression.



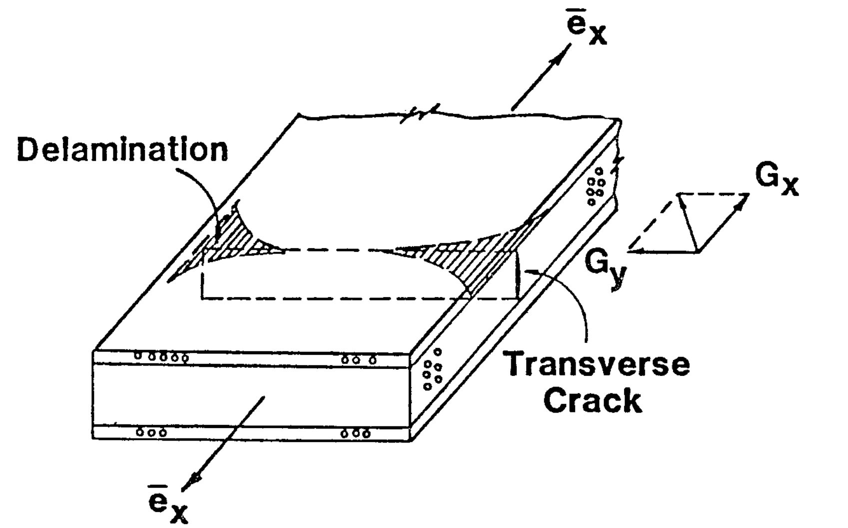
The free edge region $_z$ stress distribution......

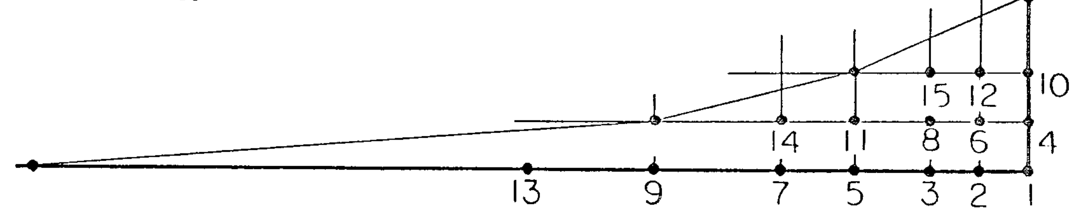


The energy release rate at mid-plane and 45/-45 plane......

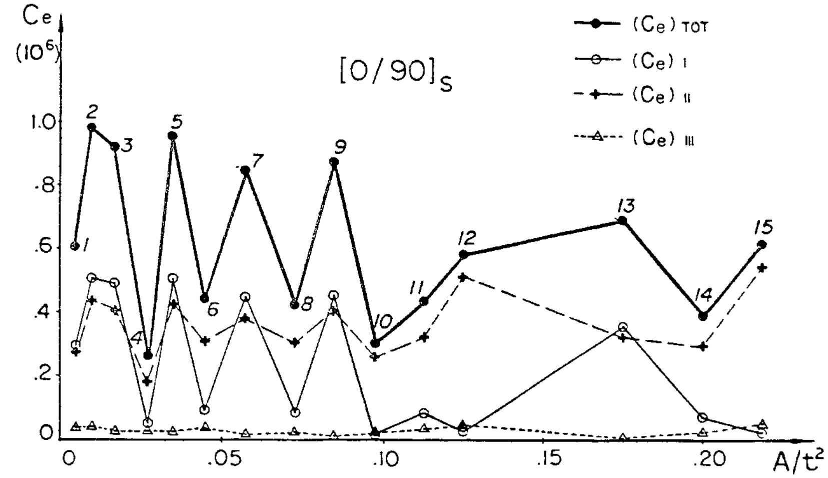
FOR TRANSVERSE CRACK/DELAMINATION INTERACTING EFFECTS......

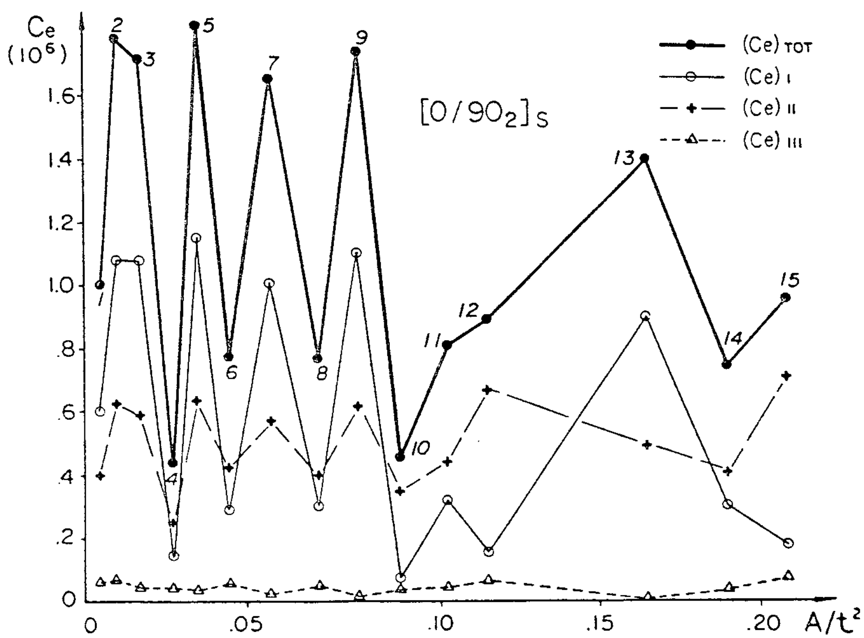
Consider the $(0/90)_s$ type laminates as an example:





90° LAYER THICKNESS EFFECT ON THE ENERGY RELEASE RATE:





A STUDY OF POLYMER MATRIX FATIGUE PROPERTIES

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Donald F. Adams

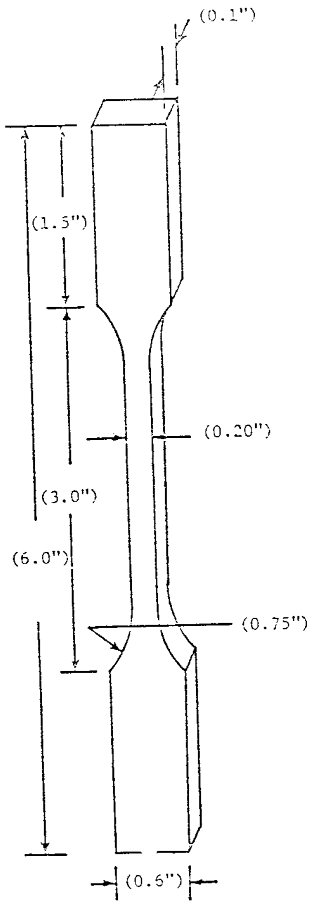
COMPOSITE MATERIALS RESEARCH GROUP
Department of Mechanical Engineering
University of Wyoming
Laramie, Wyoming 82071

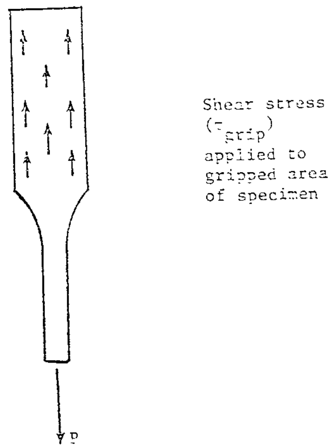
OBJECTIVES OF STUDY

1. Develop neat resin casting and testing techniques.

2. Determine fatigue response of polymer matrix materials.

3. Study any material behavior observed that would be considered atypical.





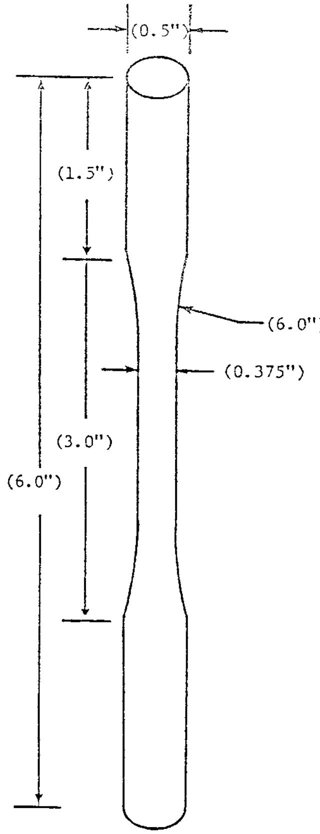
$$P = (\sigma_{ult})\ \text{(cross-sectional area of gage section)}$$

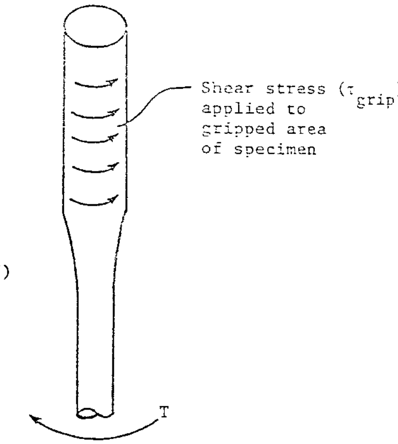
$$= (\tau_{grip})\ \text{(area of specimen gripped)}$$

$$\tau_{grip} = \frac{(\sigma_{ult})\ \text{(area of gage section)}}{\text{(area of specimen gripped)}}$$

$$= \frac{(10{,}000)\ (0.20 \times 0.1)}{2\ (0.6 \times 1.5)} = 109 \text{ psi}$$

Tension Specimen Geometry





from Reference [9] and Appendix B

$$T = \frac{\tau_{ult}\ J}{c} = \frac{(12500)\left(\frac{\pi 0.375^4}{32}\right)}{\left(\frac{0.375}{2}\right)} = 129 \text{ in-lb}$$

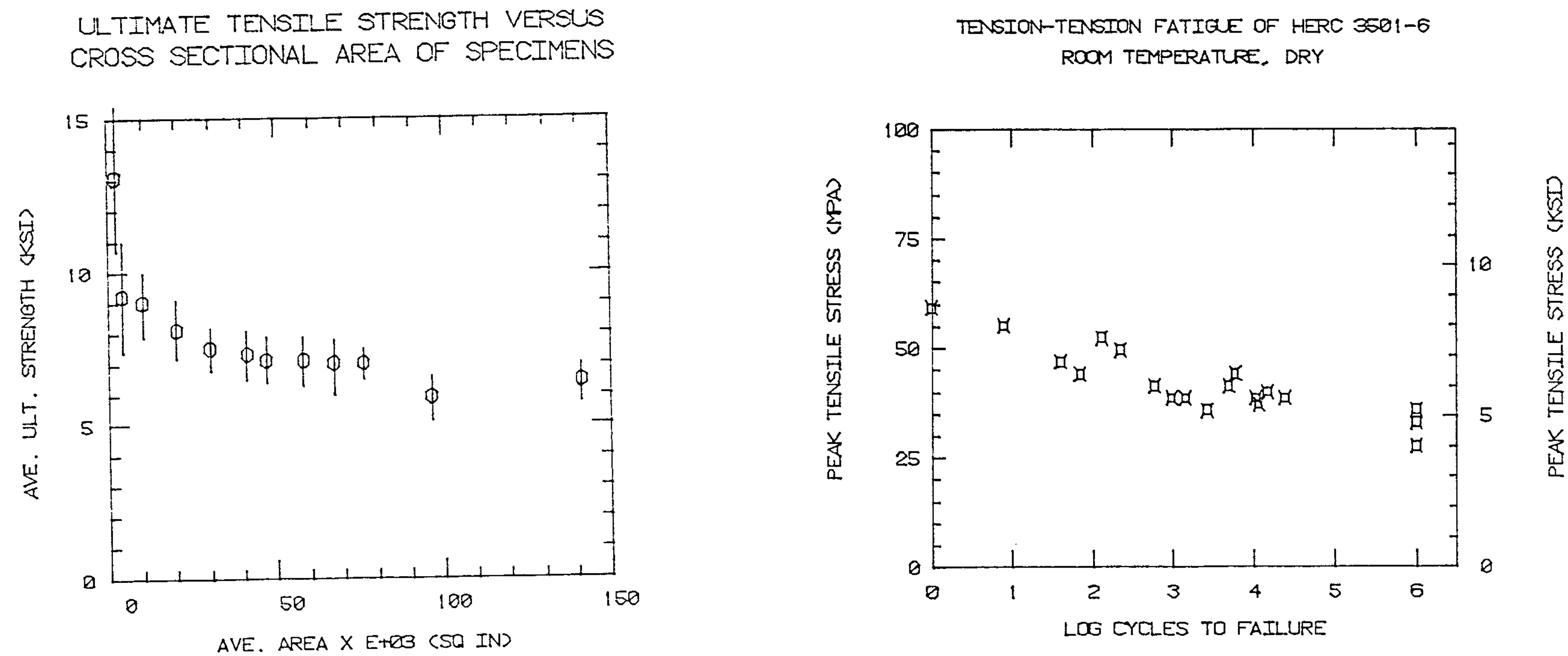
$$\Sigma M = 0 = (-T) + \tau_{grip} A_{grip} R_{grip}$$

or

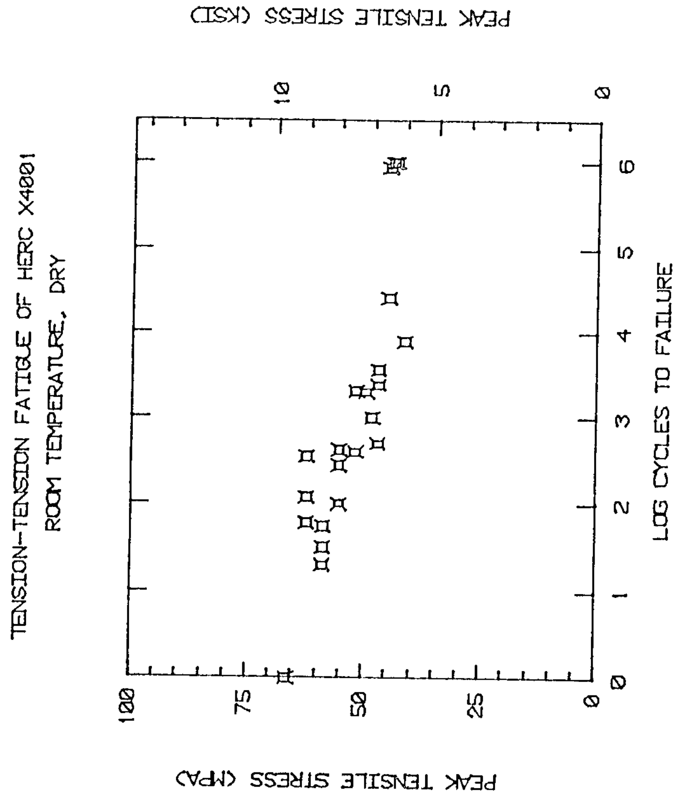
$$\tau_{grip} = \frac{T}{A_{grip} R_{grip}} = \frac{129}{2\pi(0.25)1(0.25)}$$

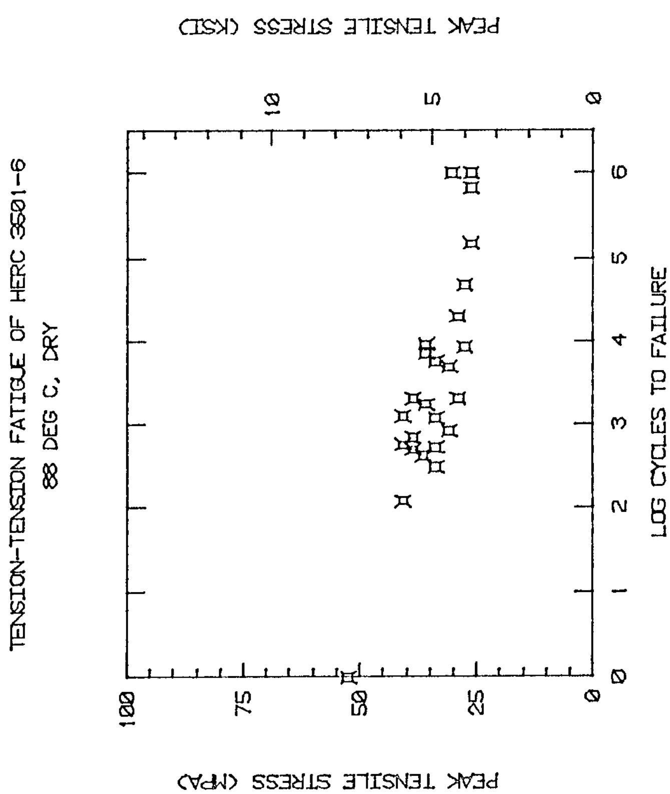
$$\tau_{grip} = 328 \text{ lb/in}^2$$

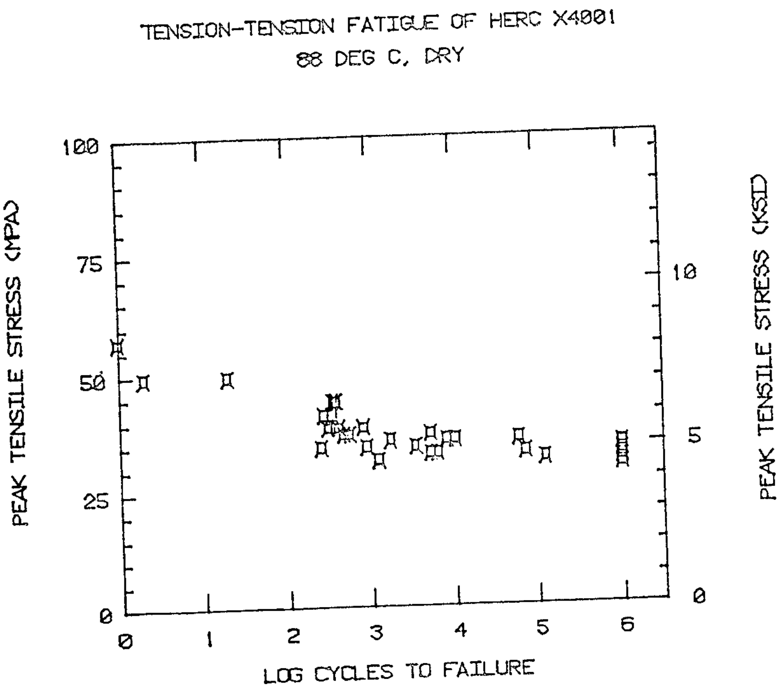
Torsion Specimen Geometry

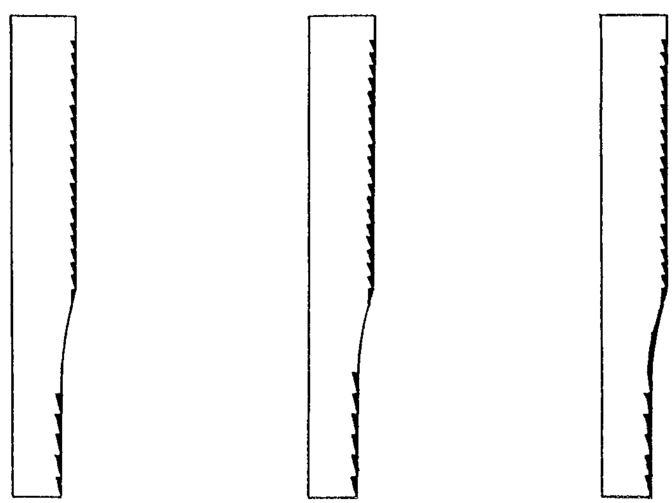


Hercules 3501-6 Neat Resin Tension Tests, Room Temperature Dry


TENSION-TENSION FATIGUE OF HERC X4001
ROOM TEMPERATURE, DRY
PEAK TENSILE STRESS (KSI)
PEAK TENSILE STRESS (MPA)
LOG CYCLES TO FAILURE

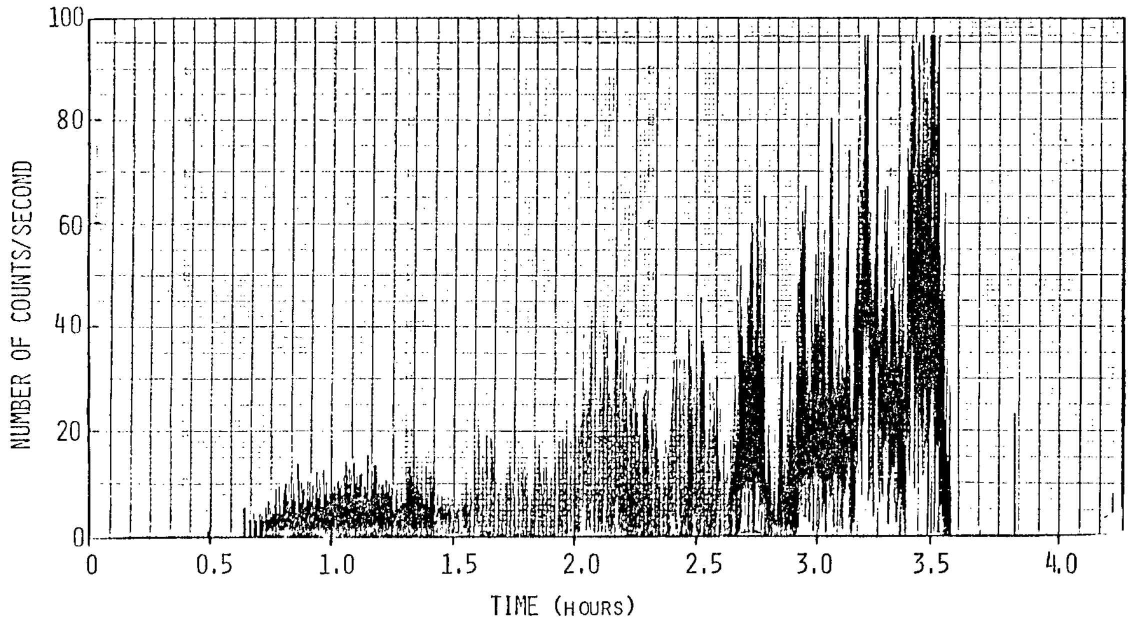

TENSION-TENSION FATIGUE OF HERC 3501-6
88 DEG C, DRY
PEAK TENSILE STRESS (KSI)
PEAK TENSILE STRESS (MPA)
LOG CYCLES TO FAILURE





a) 30 minutes b) 45 minutes c) 60 minutes

Extent of Crack Initiation and Propagation as a Function of Time Predicted by the Finite Element Analysis for the Hercules 3501-6 Epoxy Matrix Specimen (only one quadrant of specimen shown, see Figure 27).



Number of Acoustic Emission Counts per Second vs. Time for the Hercules 3501-6 Neat Epoxy. Subjected to Rapid Dryout from the Moisture-Saturated Condition.

CONCLUSIONS

1. Methods for casting neat resin specimens were developed.

2. Testing techniques were developed. However further development is needed.

3. The fatigue response of 3501-6 and 4001 is believed to be linear when plotted as peak stress versus log cycles to failure.

4. Conducting fatigue tests on moisture saturated neat resin specimens will be a difficult task.

5. The measured tensile strength of neat resins can be dependent on specimen size.

THE EFFECT OF SERVICE ENVIRONMENT ON THE MECHANICAL PROPERTIES OF COMPOSITES

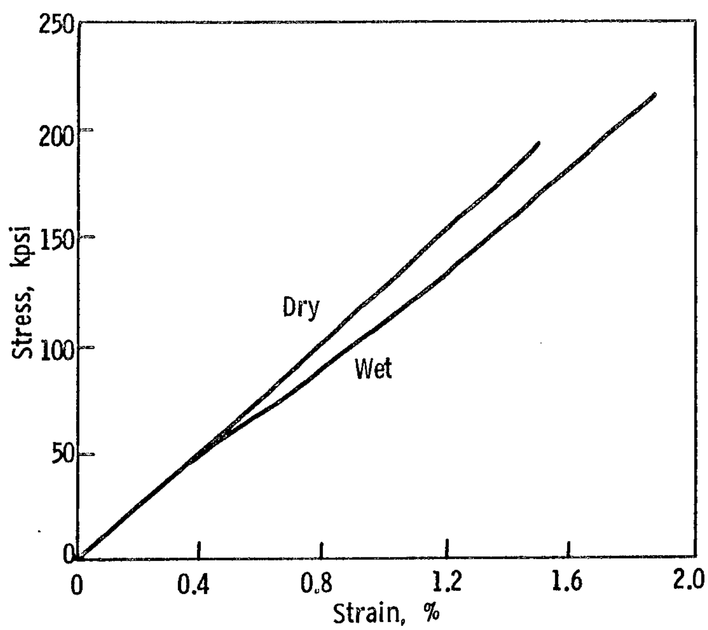
M. Roylance
W. Houghton
E. Pattie

OBJECTIVE

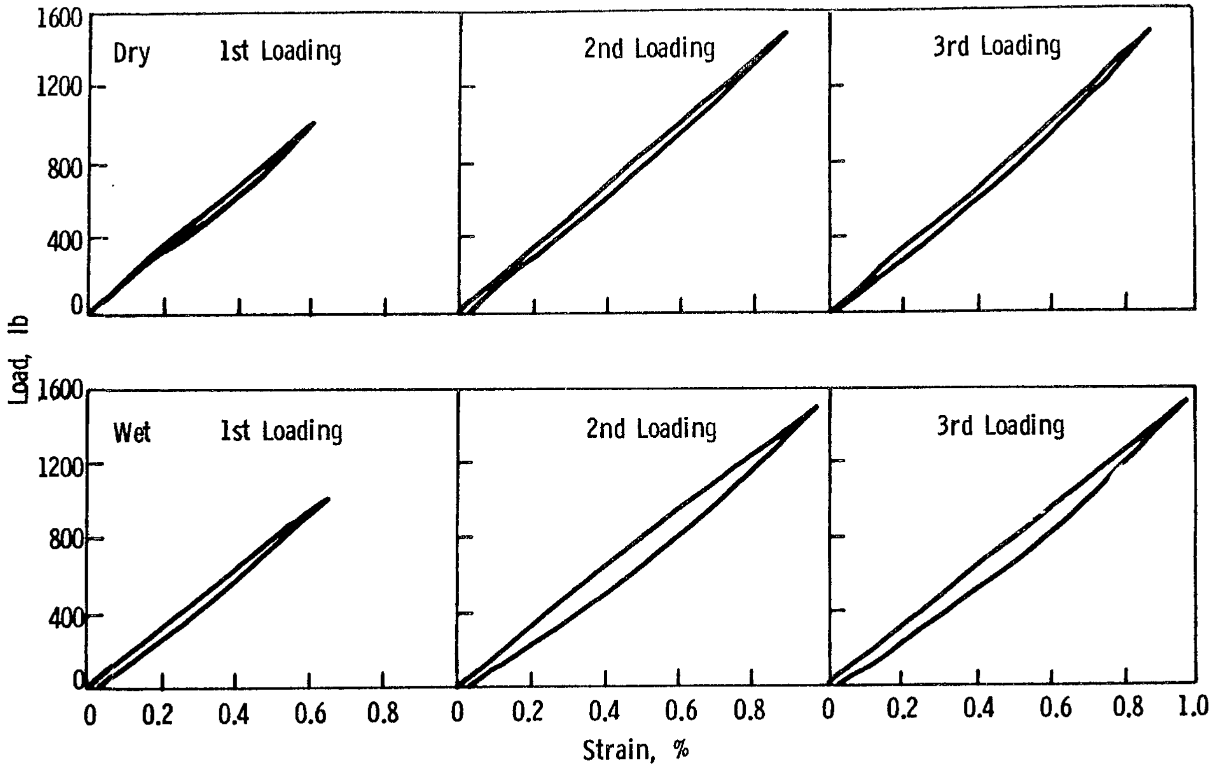
TO ASSESS THE EFFECTS OF ENVIRONMENTAL EXPOSURE ON THE DURABILITY OF VARIOUS COMPOSITE MATERIALS

SERVICE ENVIRONMENT

- ELEVATED TEMPERATURE
- ENVIRONMENTAL MOISTURE
- CYCLIC LOADING



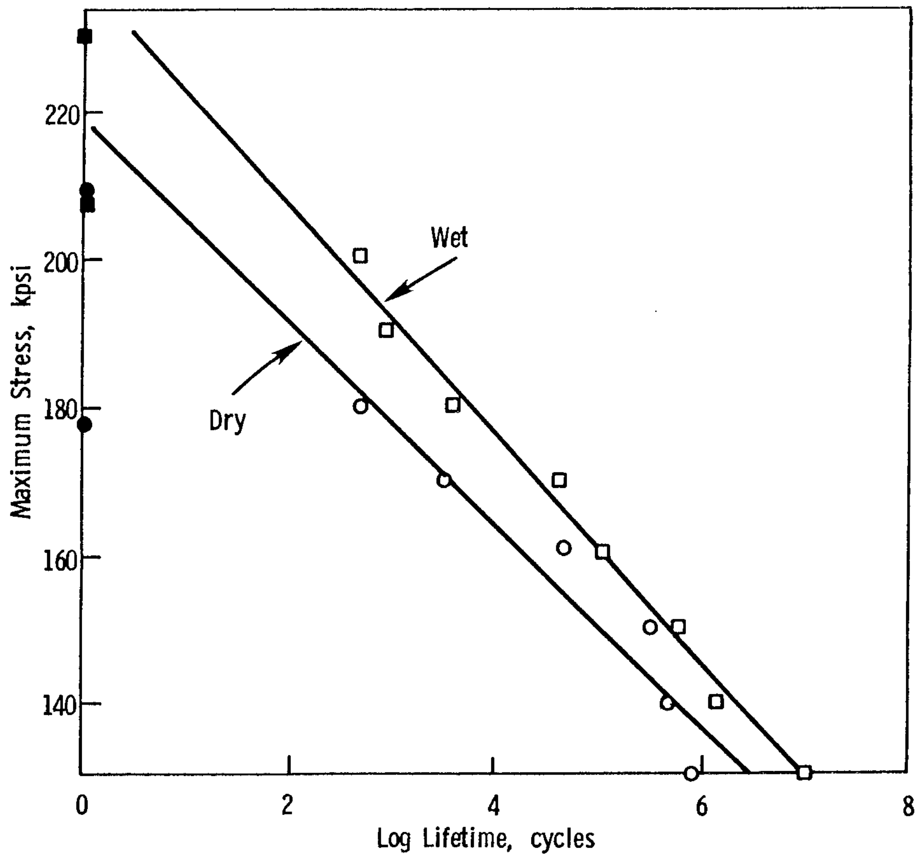
Typical Stress-Strain Curves for Wet and Dry Kevlar/934 Strip Specimens



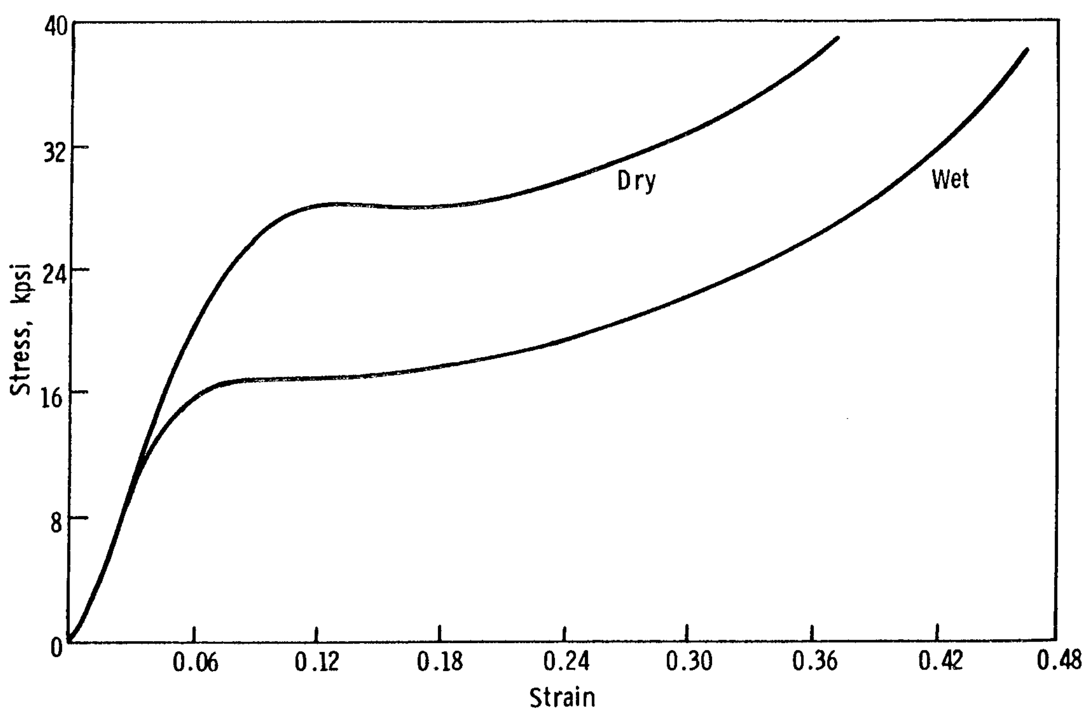
Effect of moisture on loading/unloading stress-strain curves in Kevlar/934 laminates.

TENSILE TESTS ON KEVLAR/934 LAMINATES

	Dry	Wet
UTS, kpsi	180.0 (15,6.3%)*	207.2 (14,11.3%)
Ei, Mpsi	11.12 (6,6.8%)	10.8 (5,2.5%)
Ef, Mpsi	13.0 (6,7.4%)	13.0 (5,4.4%)
ϵf,%	1.6 (6,0.5%)	1.8 (5,4.5%)



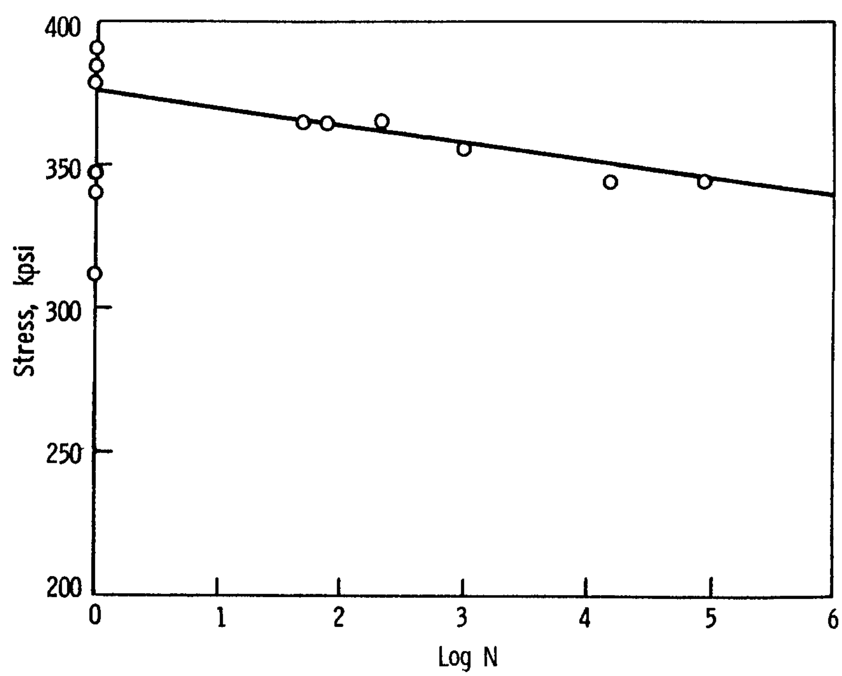
Tensile S-N Curves for Wet and Dry Kevlar/934 Strip Specimens.
Solid Symbols Indicate Single Cycle Strengths.



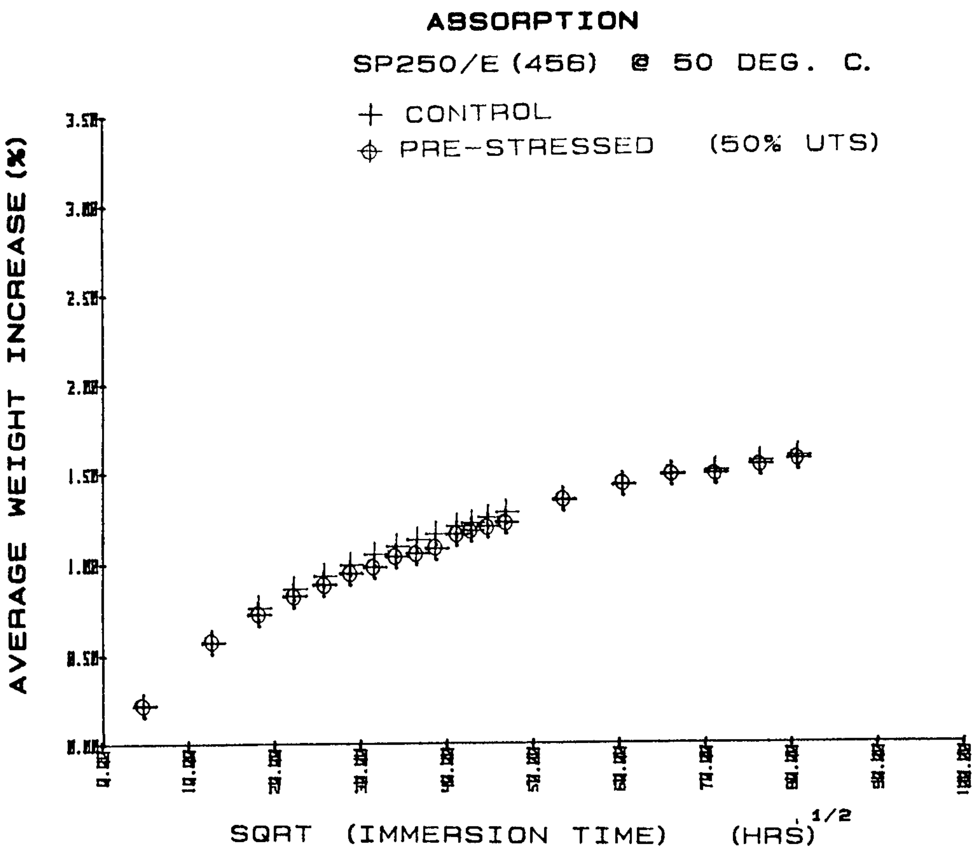
Typical Compression Stress-Strain Curves for Wet and Dry Resin

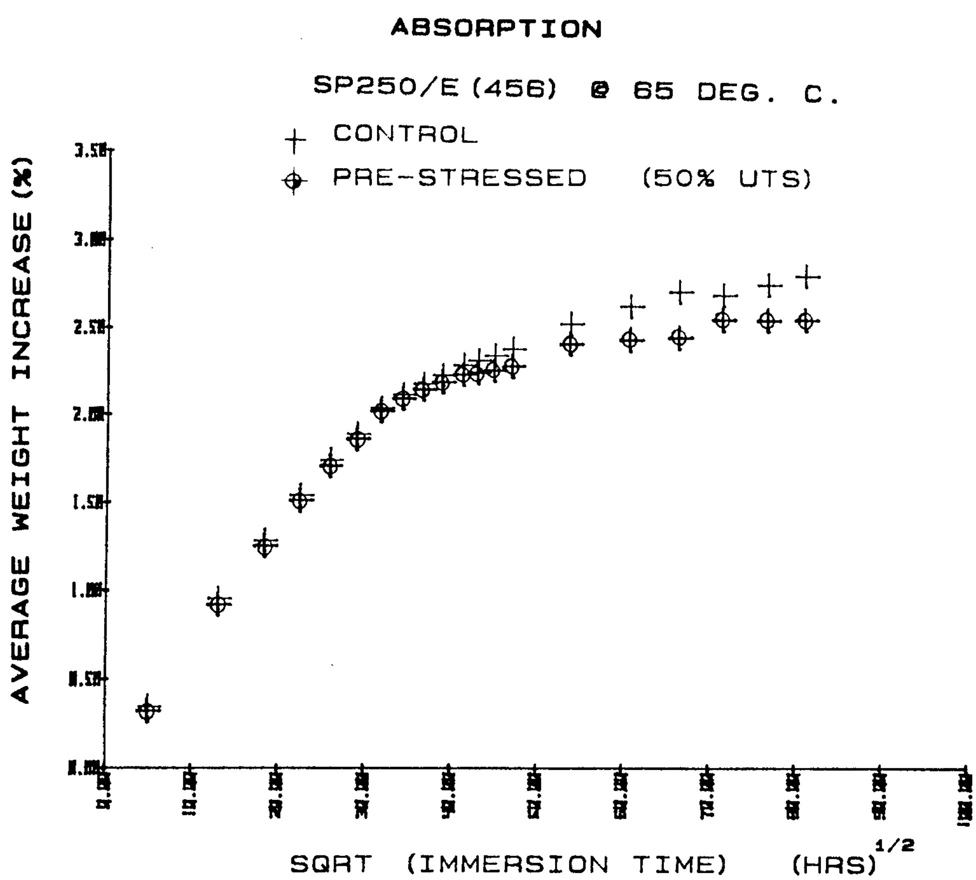
COMPRESSION TESTS ON 934 EPOXY.

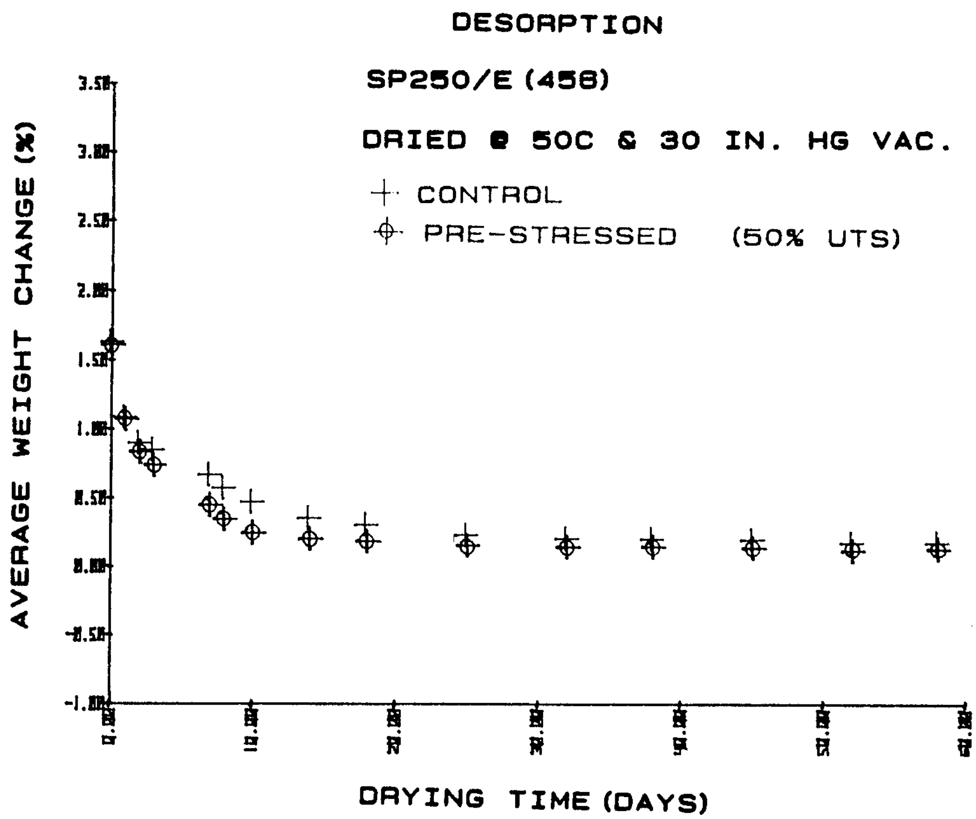
	Wet	Dry
Ei, kpsi	362 (6,2.6%)	393 (6,4.1%)
σy, kpsi	17.4 (6,1.1%)	29.4 (6,2.7%)
ϵy, %	9.4 (6,4.6%)	12.3 (6,3.3%)
σf, kpsi	39.1 (6,7.9%)	39.7 (6,14.3%)
ϵf, %	47.0 (6,2.5%)	35.6 (6,25.8%)

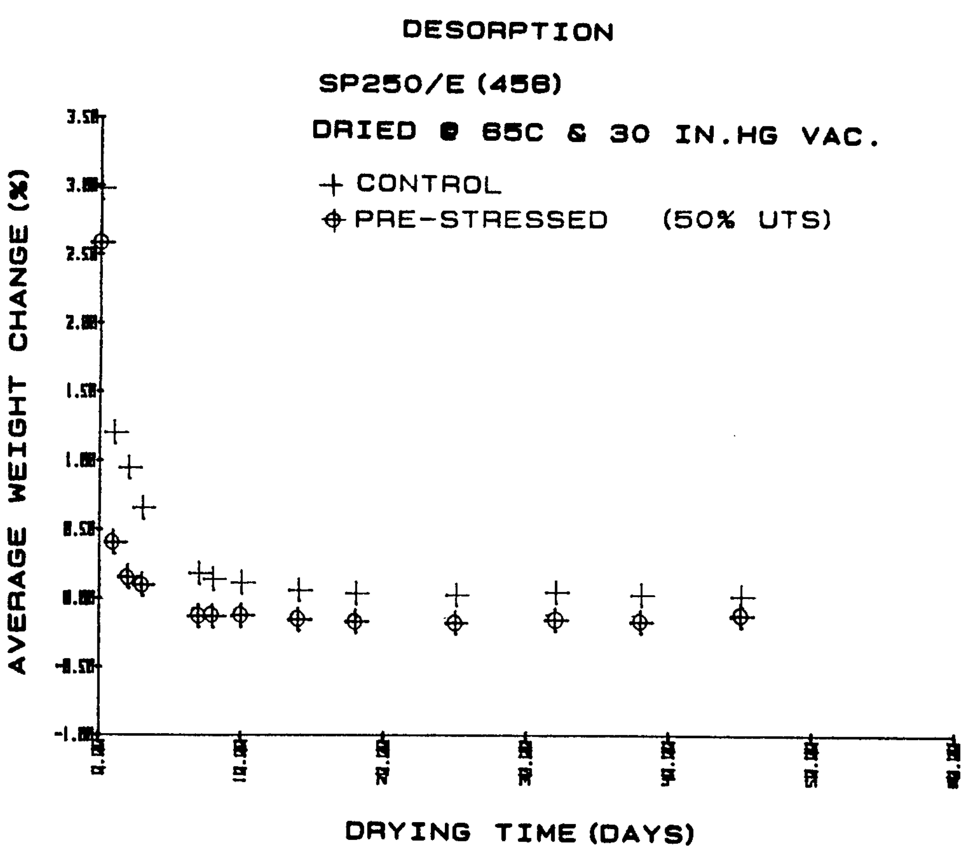


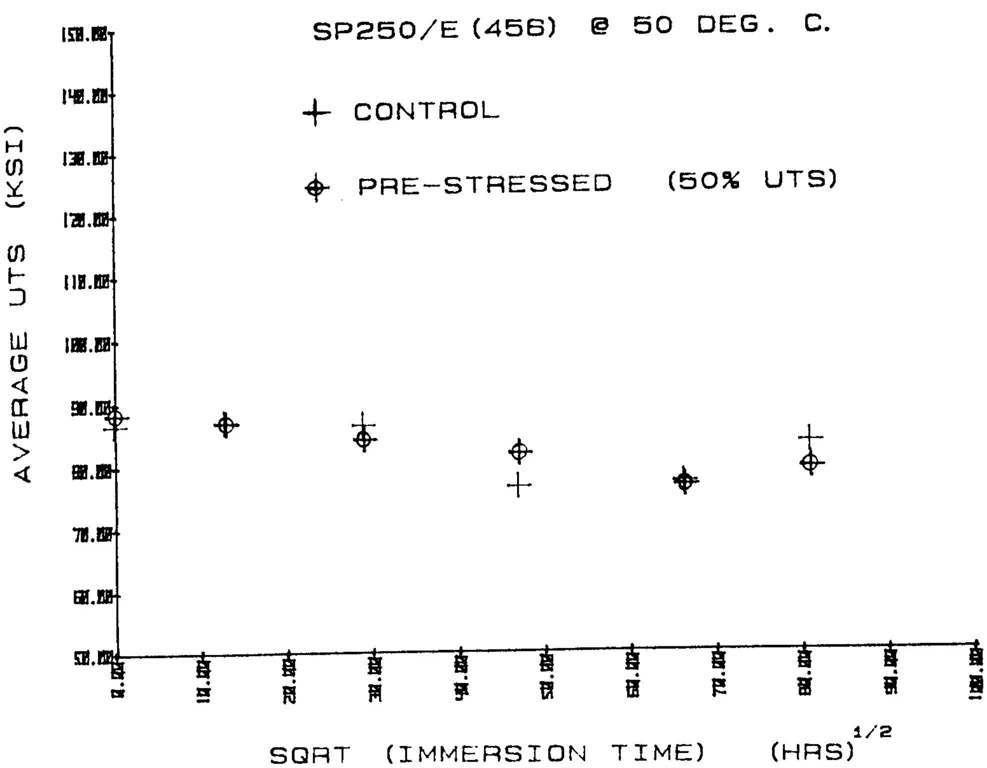
Tensile Fatigue Lifetimes for Dry Kevlar Yarn

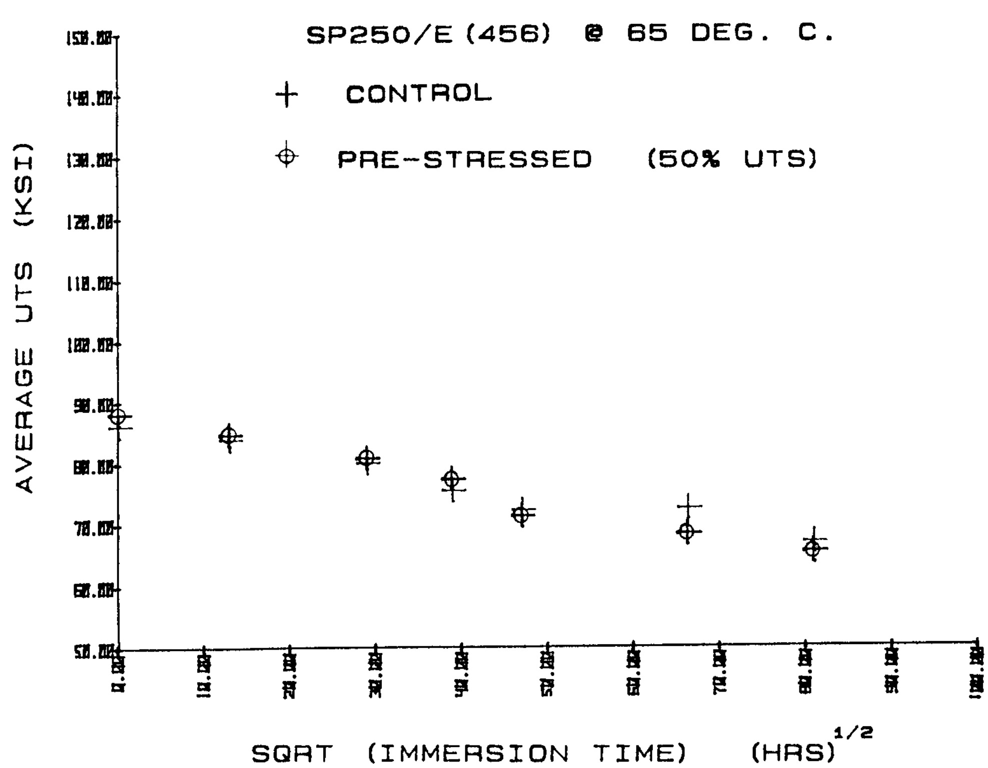

ABSORPTION
SP250/E (456) @ 50 DEG. C.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE WEIGHT INCREASE (%)
SQRT (IMMERSION TIME) (HRS)$^{1/2}$

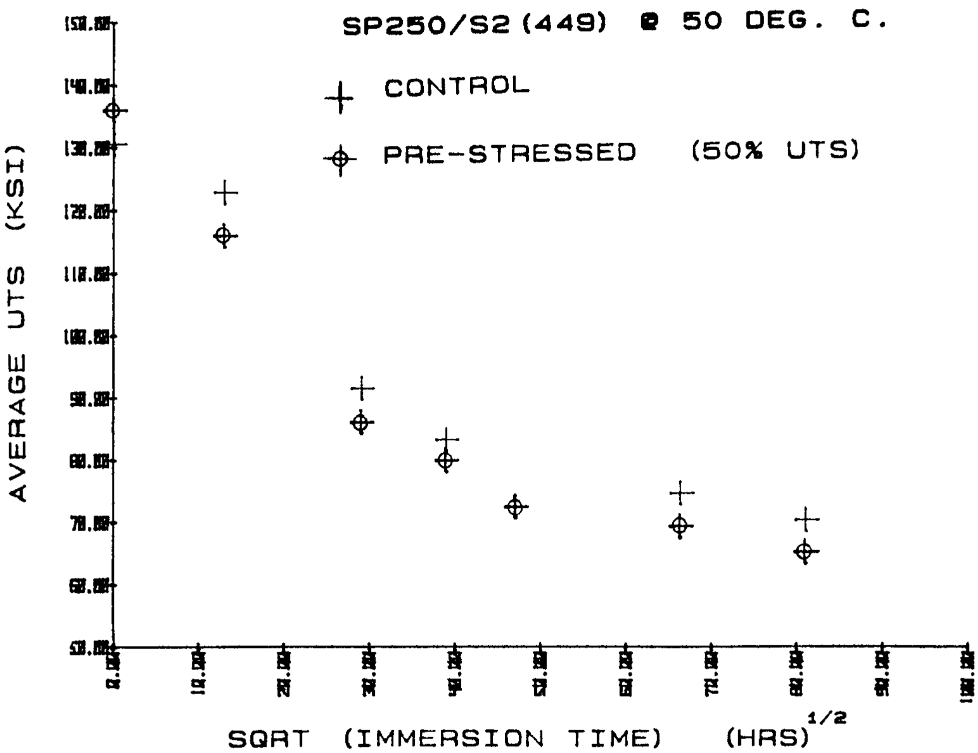

ABSORPTION
SP250/E (456) @ 65 DEG. C.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE WEIGHT INCREASE (%)
SQRT (IMMERSION TIME) (HRS)$^{1/2}$

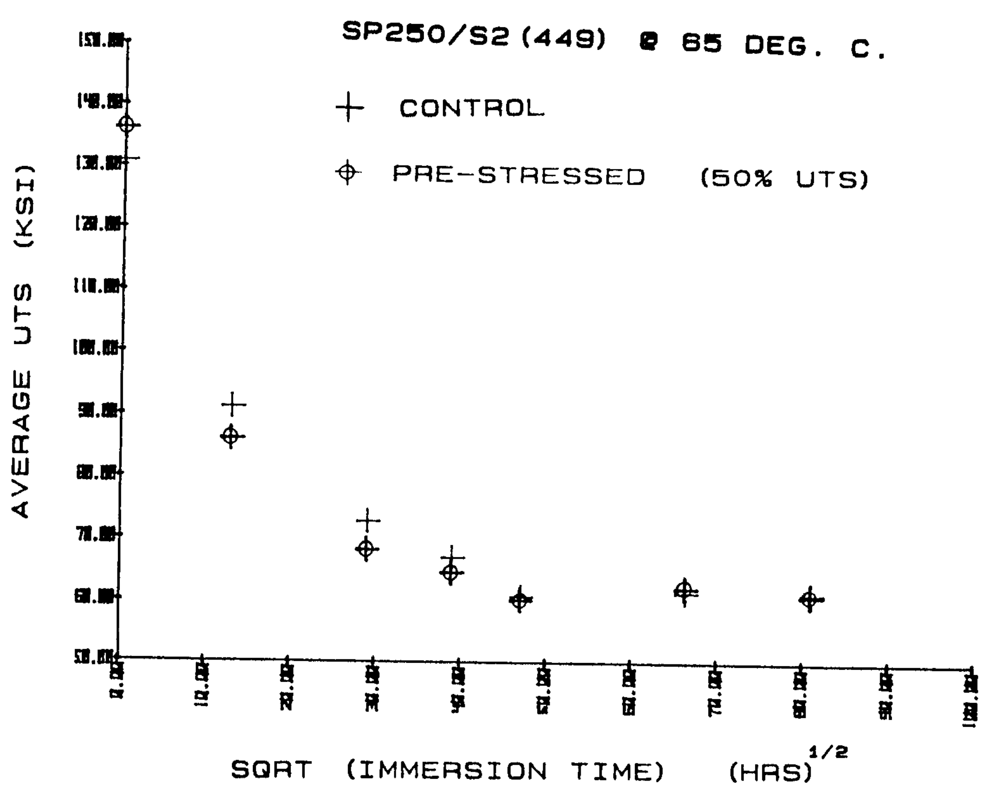

DESORPTION
SP250/E (456)
DRIED @ 50C & 30 IN. HG VAC.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE WEIGHT CHANGE (%)
DRYING TIME (DAYS)

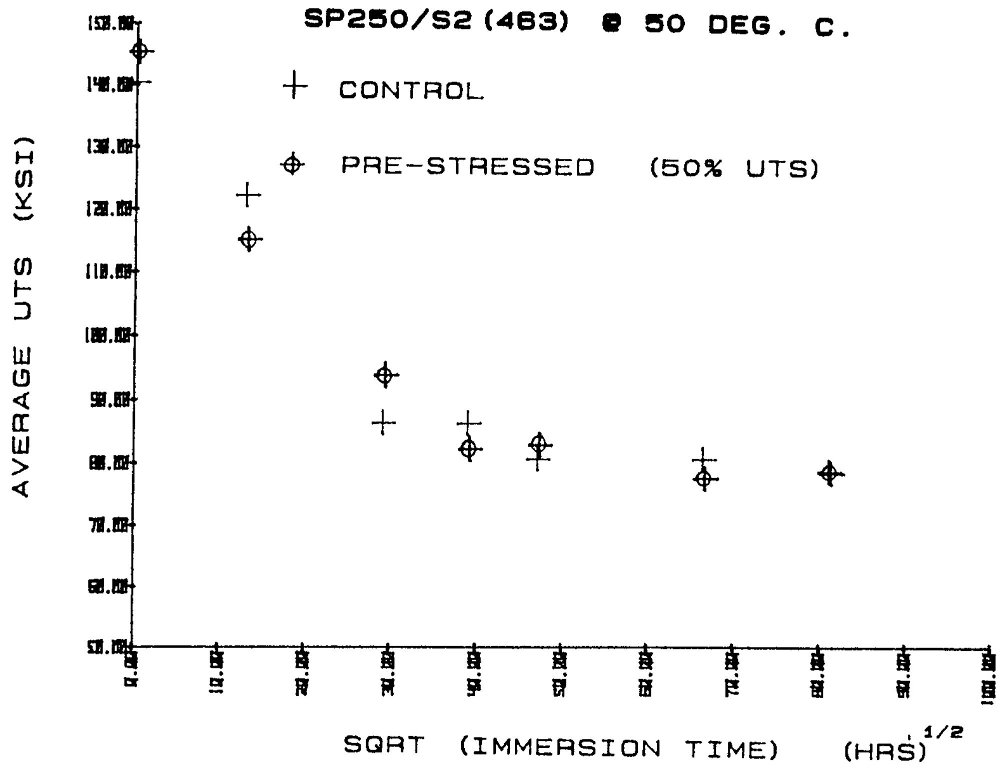

DESORPTION
SP250/E (456)
DRIED @ 65C & 30 IN.HG VAC.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE WEIGHT CHANGE (%)
DRYING TIME (DAYS)

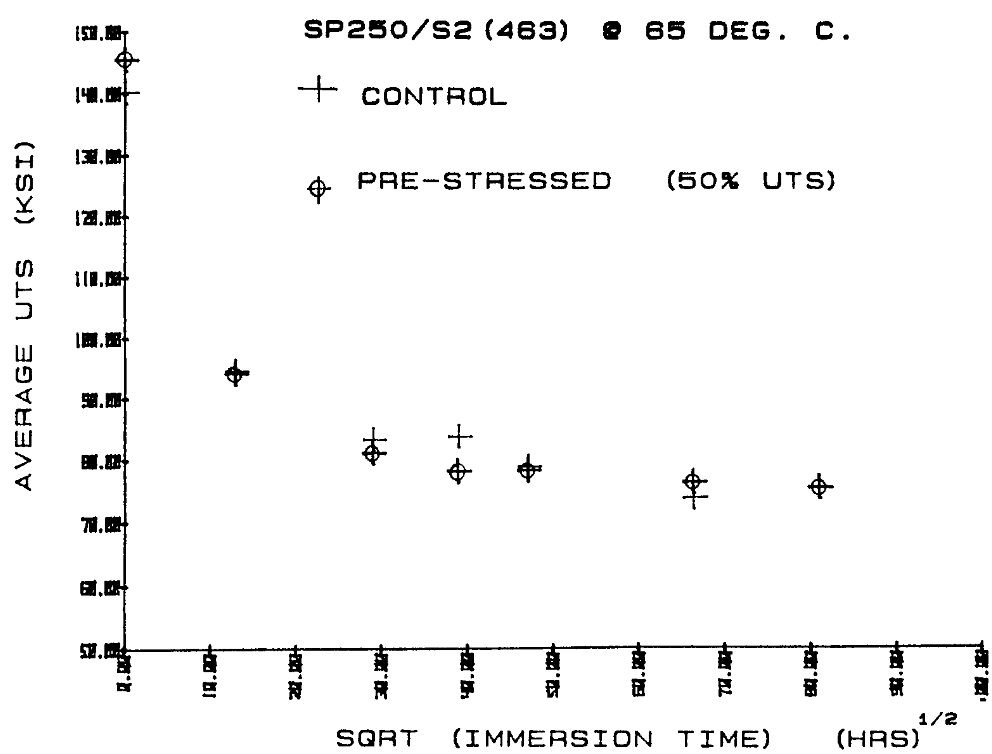

SP250/E (456) @ 50 DEG. C.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE UTS (KSI)
SQRT (IMMERSION TIME) (HRS)^1/2


SP250/E (456) @ 65 DEG. C.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE UTS (KSI)
SQRT (IMMERSION TIME) (HRS)^1/2


SP250/S2 (449) @ 50 DEG. C.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE UTS (KSI)
SQRT (IMMERSION TIME) (HRS)1/2


SP250/S2 (449) @ 65 DEG. C.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE UTS (KSI)
SQRT (IMMERSION TIME) (HRS)1/2


SP250/S2 (463) @ 50 DEG. C.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE UTS (KSI)
SQRT (IMMERSION TIME) (HRS)$^{1/2}$


SP250/S2 (463) @ 65 DEG. C.
CONTROL
PRE-STRESSED (50% UTS)
AVERAGE UTS (KSI)
SQRT (IMMERSION TIME) (HRS)$^{1/2}$

CONCLUSIONS

- ABSORBED MOISTURE INCREASES THE STRENGTH OF UNIDIRECTIONAL KEVLAR/EPOXY BY APPROXIMATELY 10%
- ABSORBED MOISTURE DECREASES THE STRENGTH OF 0/90 S-GLASS/EPOXY BY 44 TO 46%, DEPENDING UPON FIBER SURFACE TREATMENT
- ABSORBED MOISTURE DECREASES THE STRENGTH OF 0/90 E-GLASS/EPOXY BY ONLY 4%, RESULTING IN WET STRENGTH COMPARABLE TO S-GLASS COMPOSITES

Characterization of Resin Matrix Composites and the Influence of Environmental Factors on Them

by

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For Presentation at the

Ninth Annual Mechanics
of Composites Review

Dayton, Ohio

October 24-26, 1983

This work supported jointly by NASA/AFOSR

WHY STUDY SMALL AMPLITUDE DYNAMIC BEHAVIOR ?

LARGE AMPLITUDES GIVE RISE TO :

- ◇ NONLINEAR PHENOMENA
- ◇ FAILURE PROCESSES
- ◇ PERMANENT STRUCTURAL CHANGES
- ◇ EXCESSIVE HEAT GENERATION

SMALL AMPLITUDES :

- ◇ ARE GENERALLY LINEAR VISCOELASTIC
- ◇ GENERATE LITTLE HEAT
- ◇ CAUSE LIMITED STRUCTURAL CHANGES

SMALL AMPLITUDE DYNAMIC MECHANICAL SPECTROSCOPY CAN BE USED FOR :

MATERIALS CHARACTERIZATION, SUCH AS COMPARISON OF FABRICATION RESULTS, THERMAL HISTORY, RESIDUAL SOLVENT, AND ALTERATIONS OF STRUCTURE DUE TO PROCESSING

DETERMINING MATERIAL PARAMETER INPUTS FOR MICROMECHANICS AND FAILURE MODELING STUDIES

PRELIMINARY EVALUATION OF CANIDATE RESINS FOR ENHANCED TOUGHNESS AND DAMAGE TOLERANCE

HIGHLY RESIN SENSITIVE CHARACTERIZATION OF IN-SITU RESINS

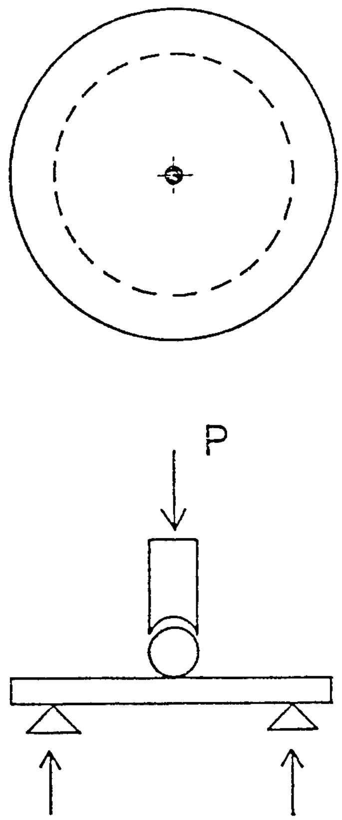


Figure 1.

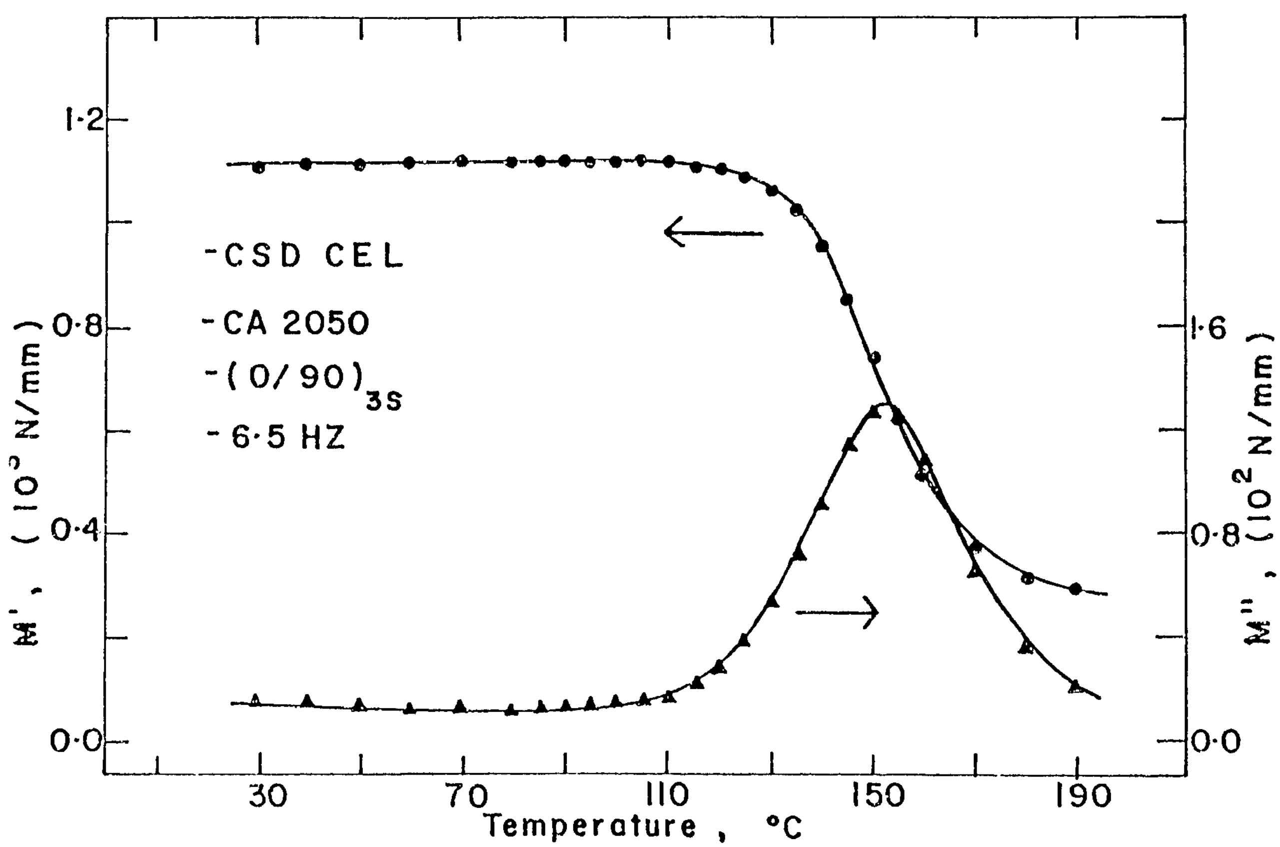


Figure 2.

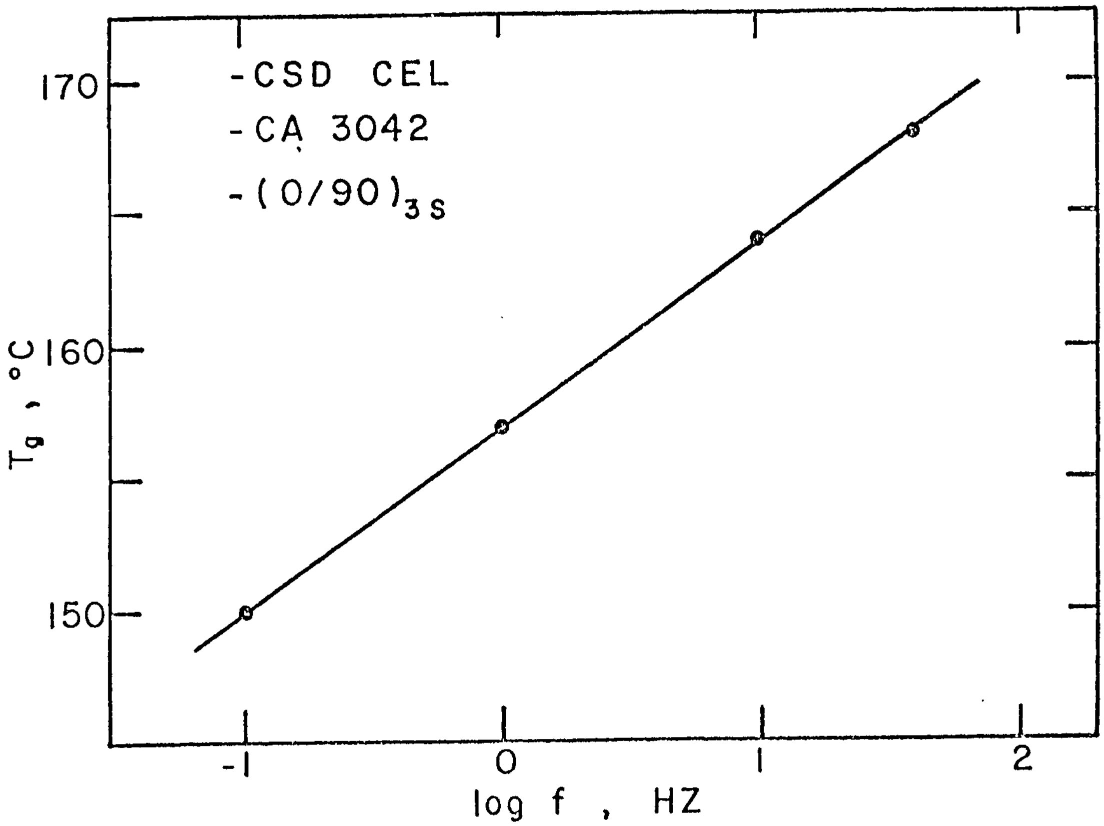


Figure 3.

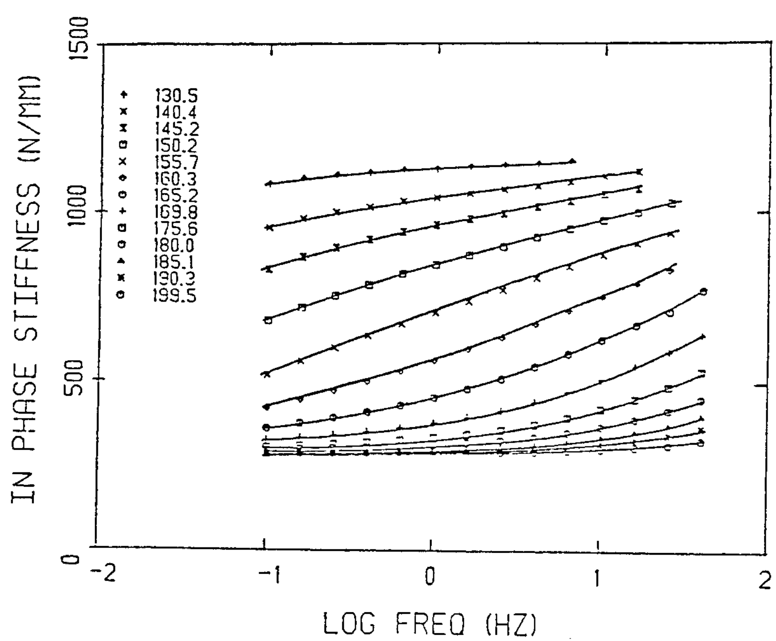


Figure 4.

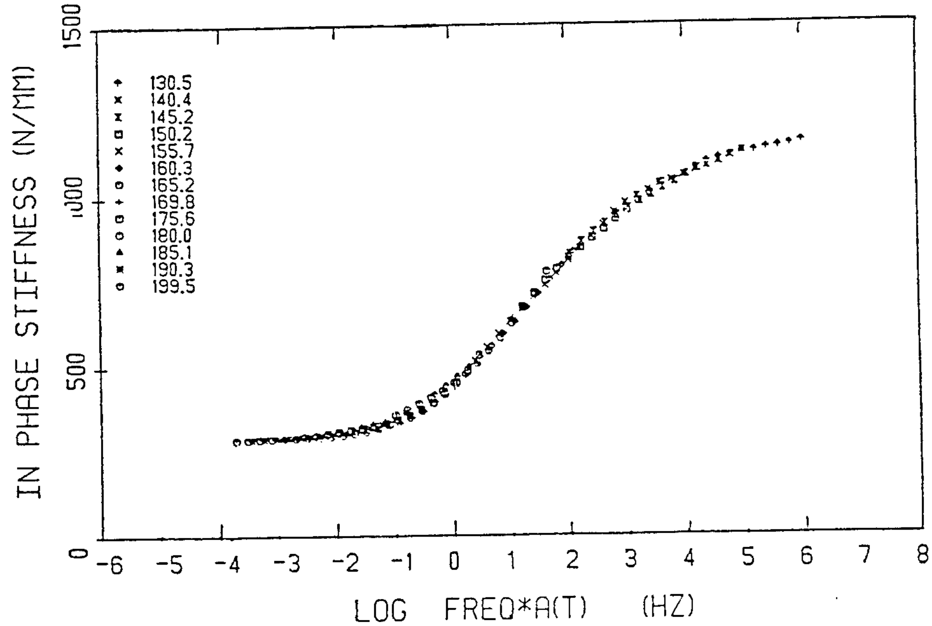


Figure 5.

6. Master Curve of Storage Compliance--Linear Plot

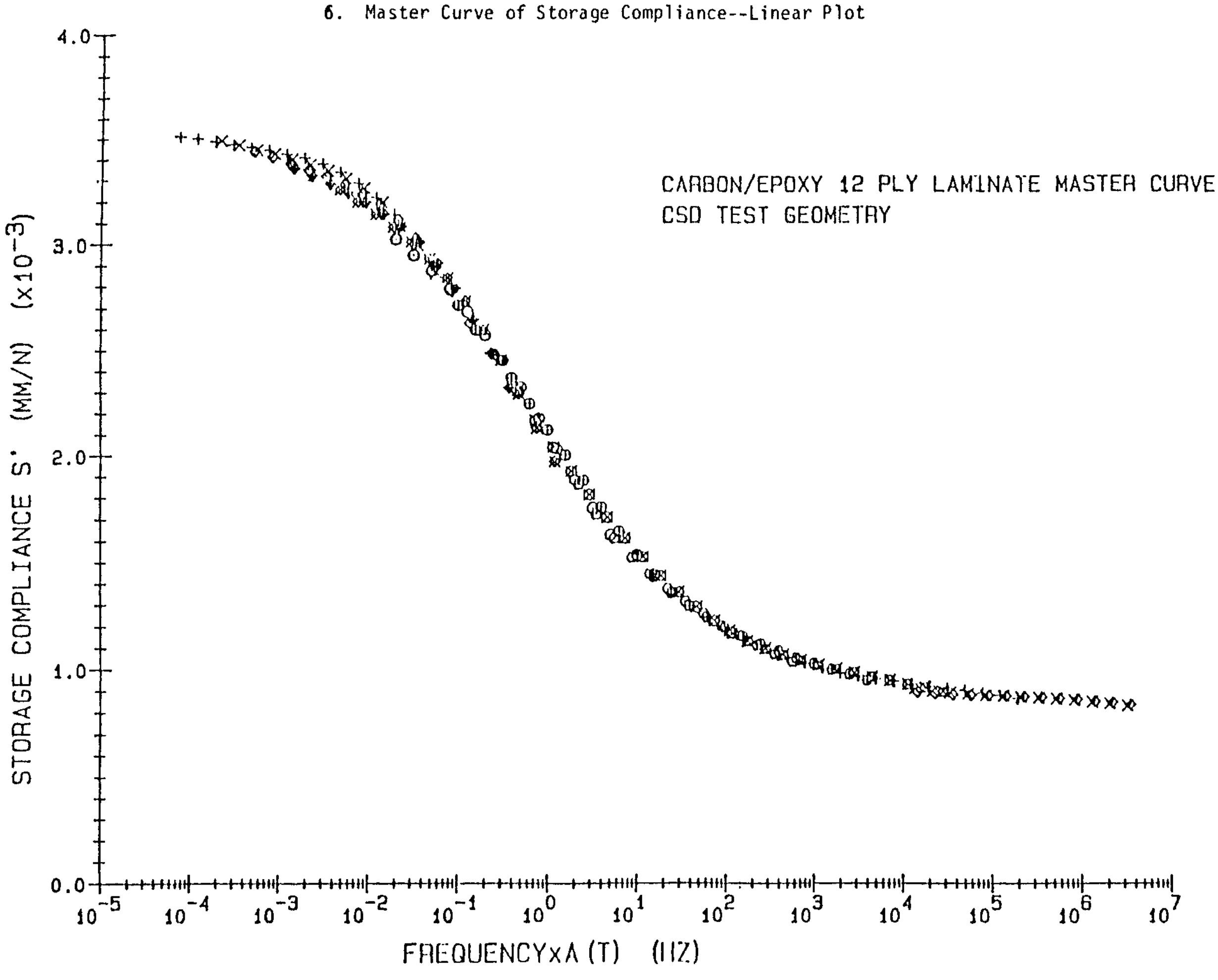


Figure 6.

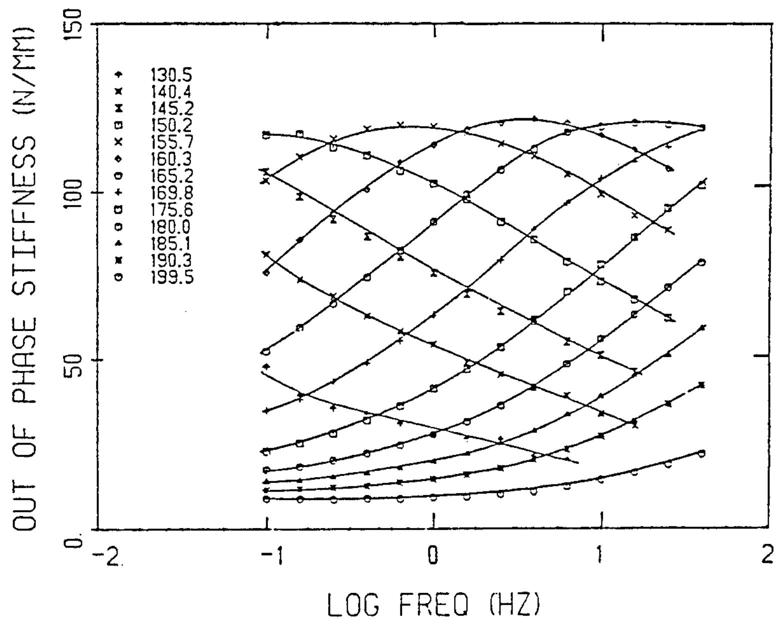


Figure 7.

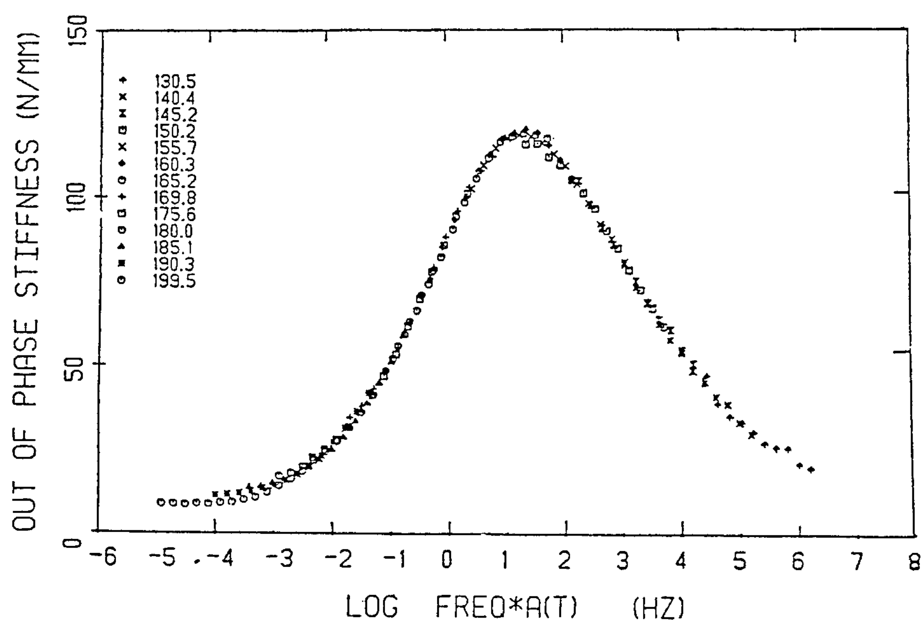


Figure 8.

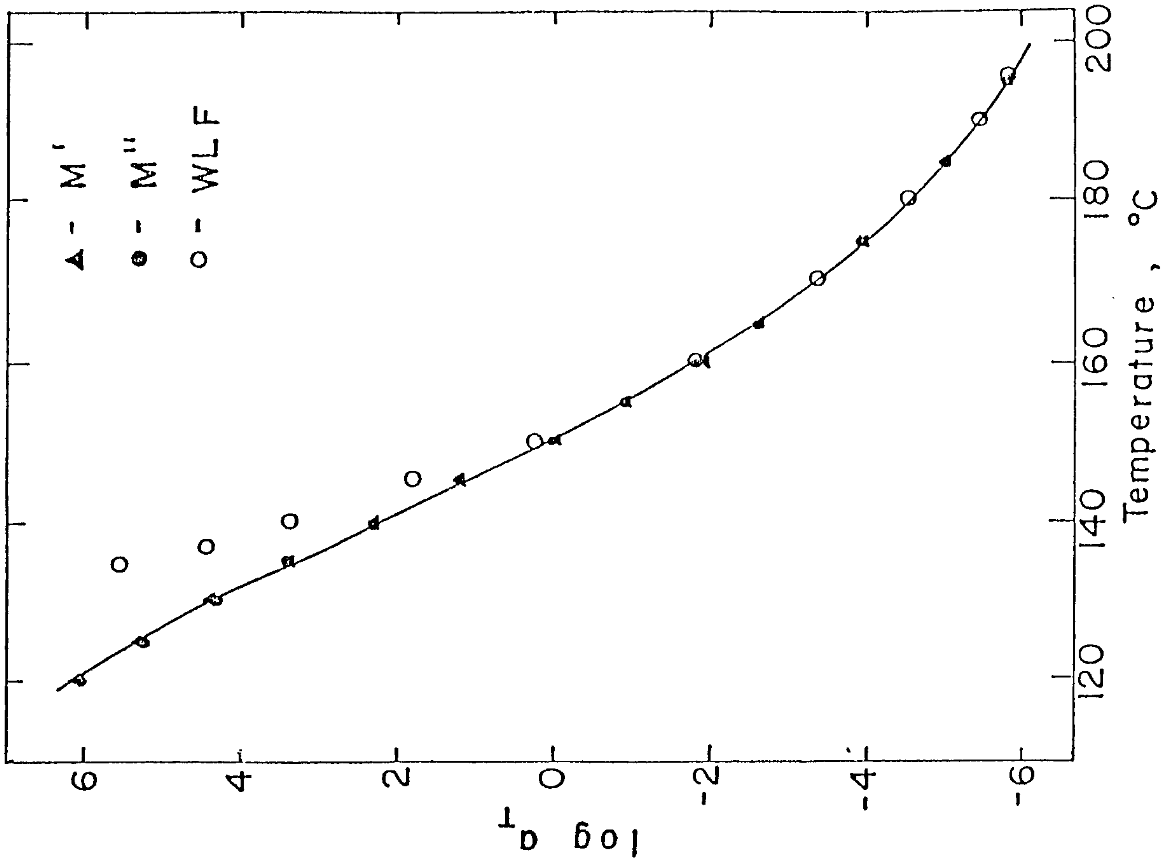


Figure 9.

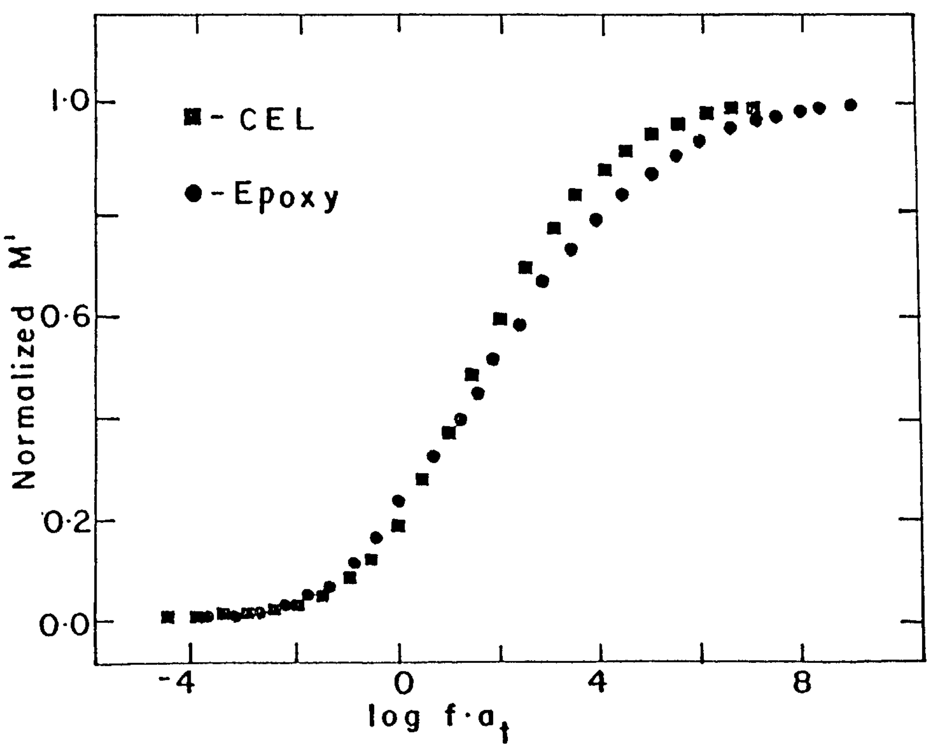


Figure 10.

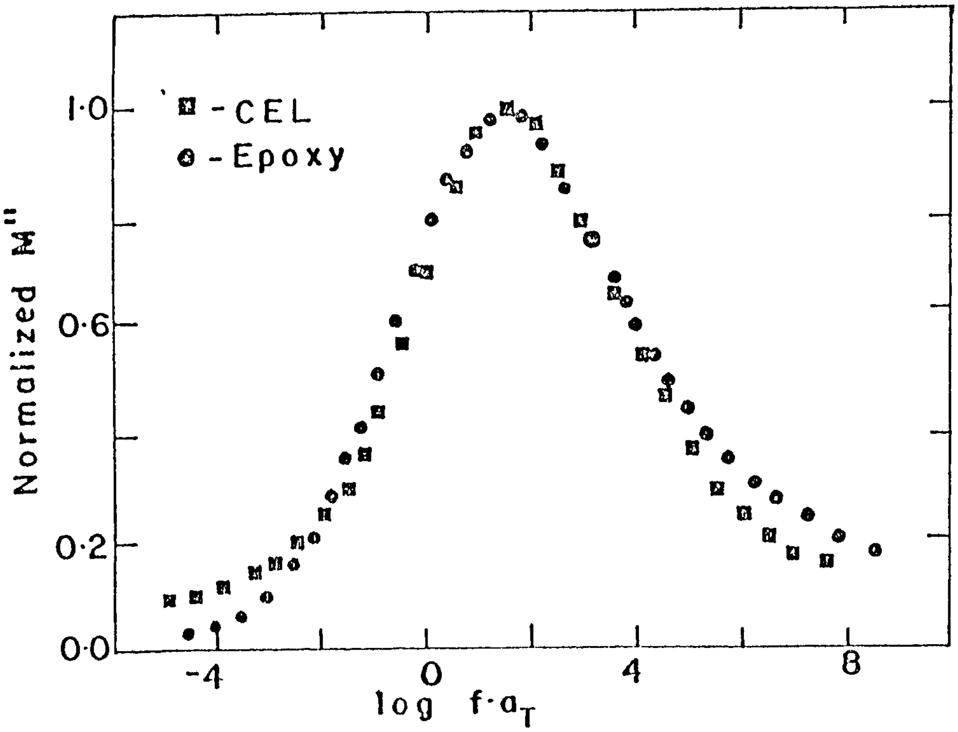


Figure 11.

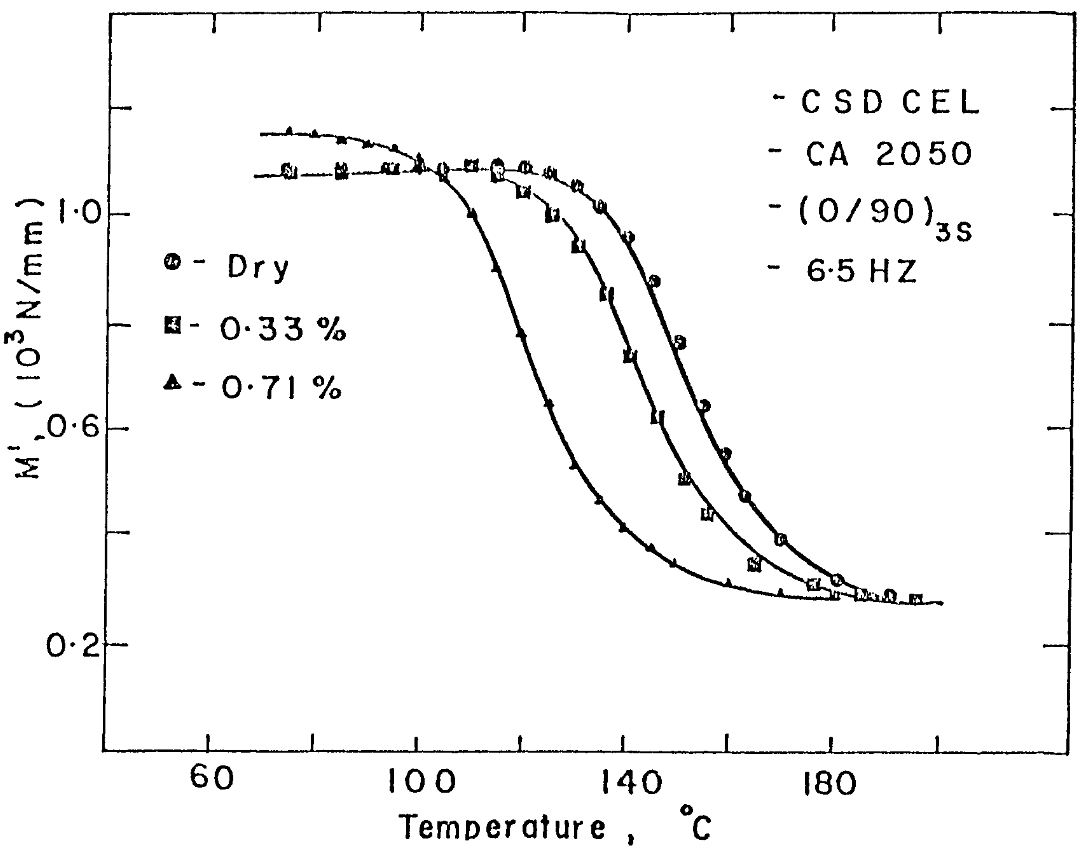


Figure 12.

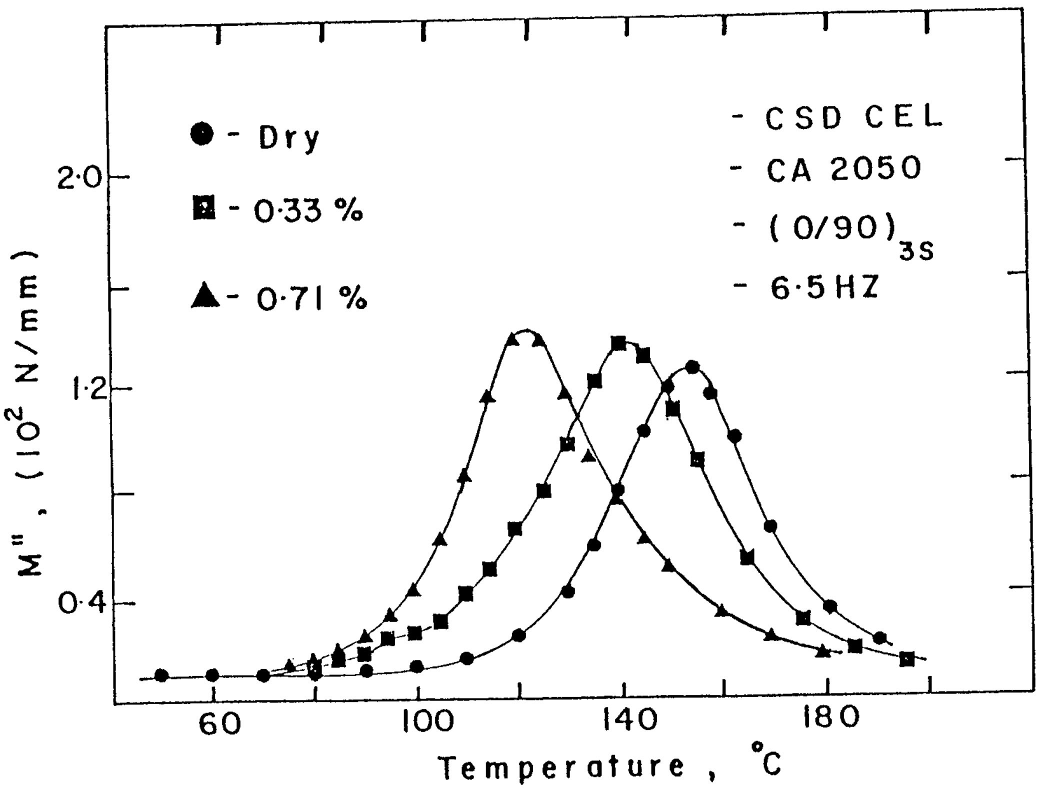


Figure 13.

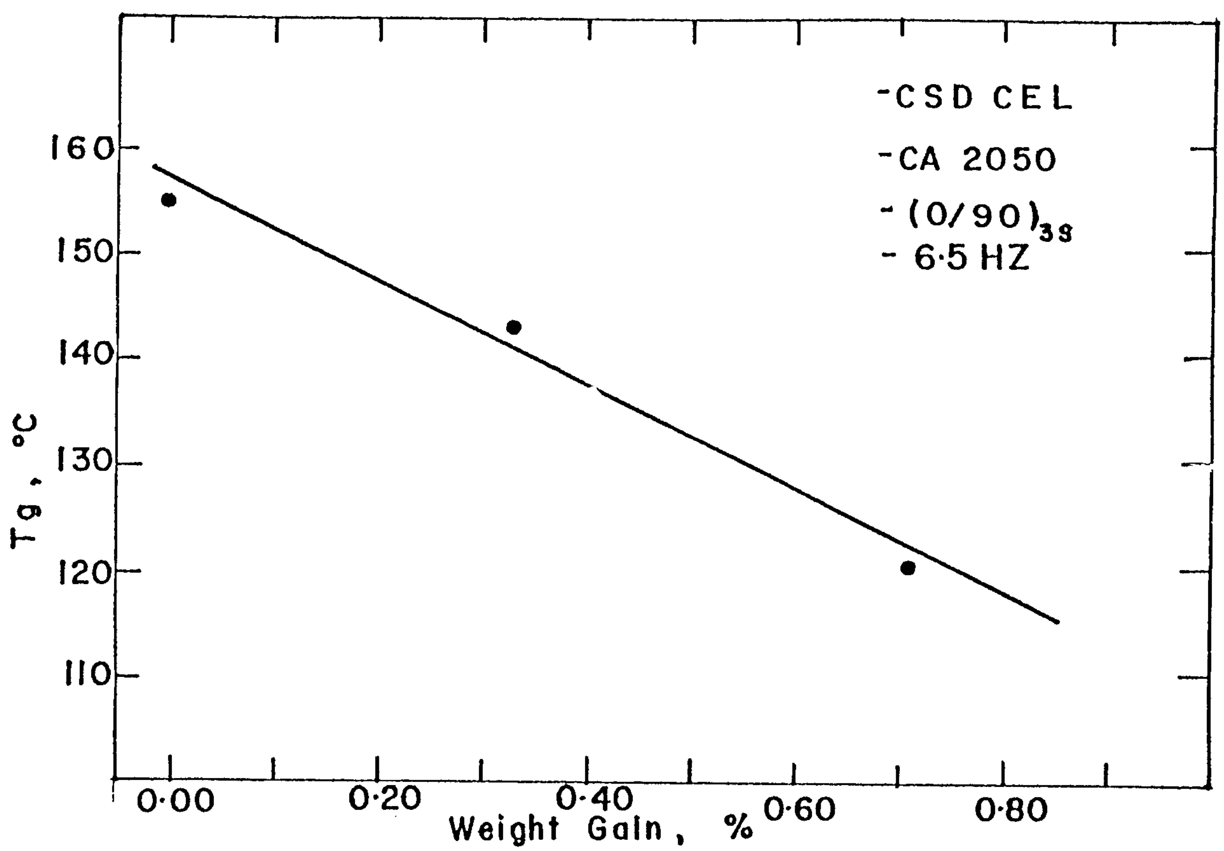


Figure 14.

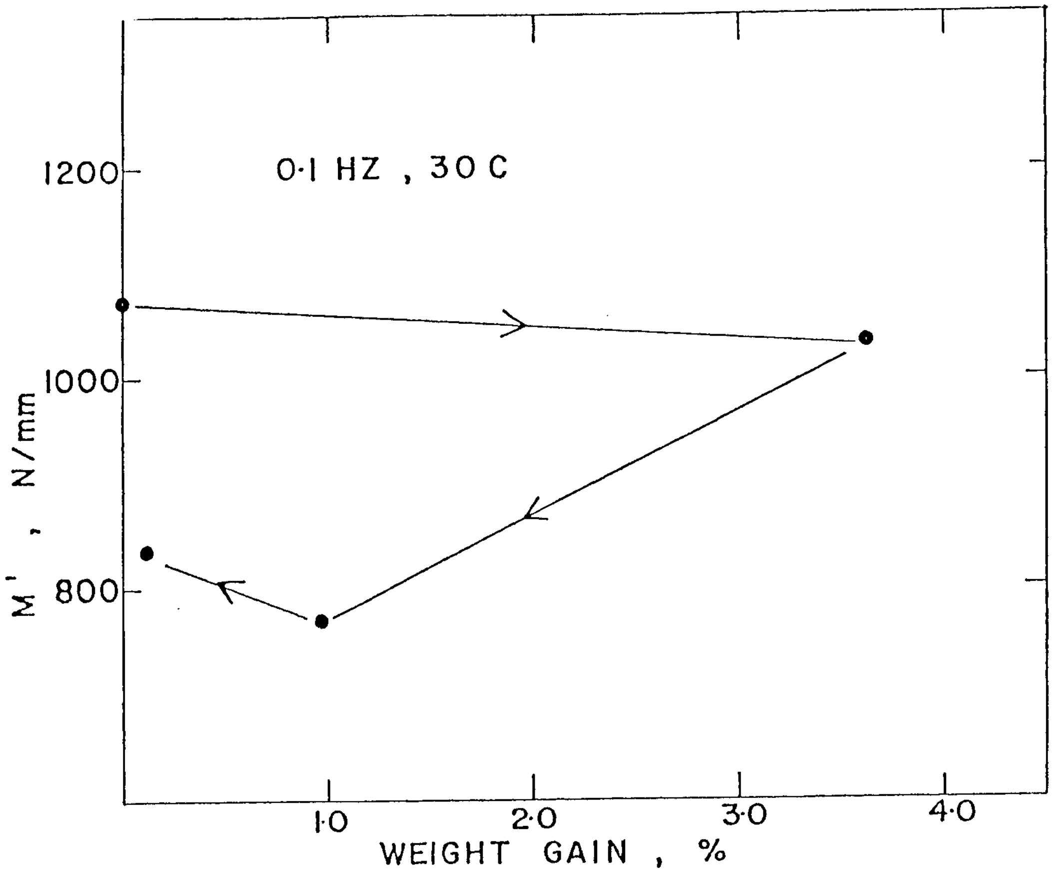


Figure 15. Summary Comparison of Carbon/Epoxy Specimen Stiffness (normalized to dry sample thickness)

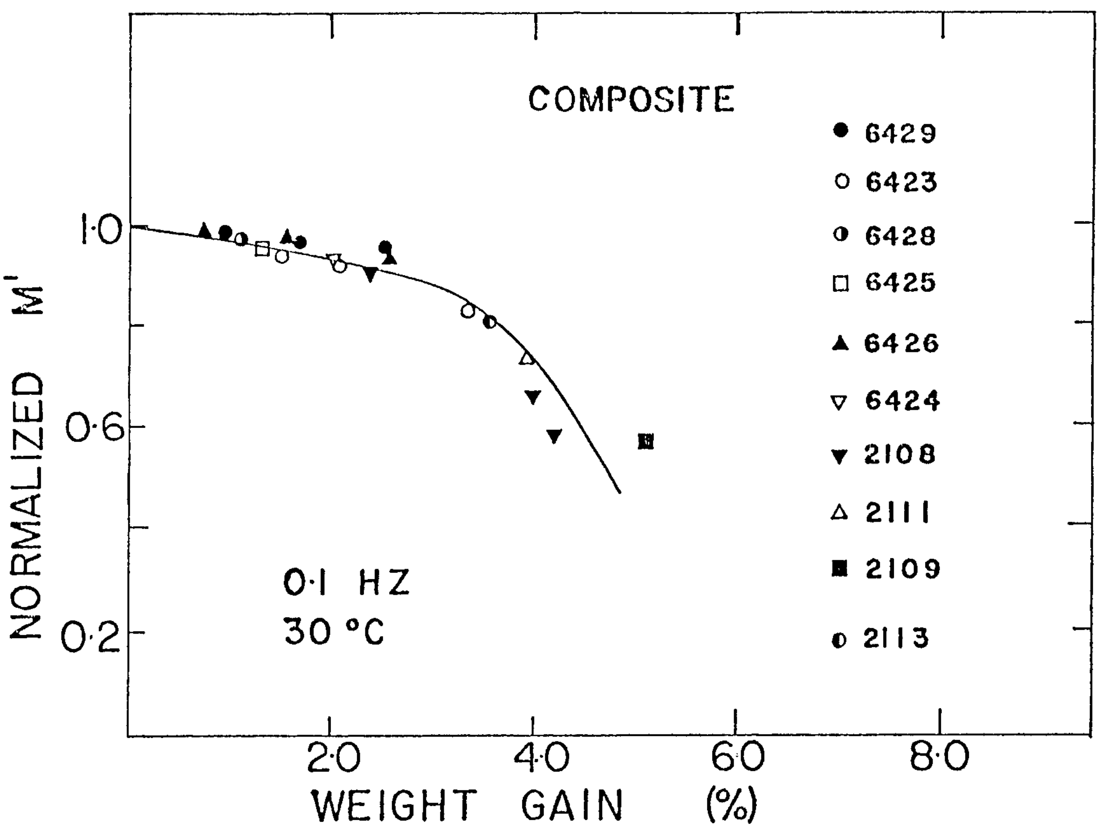


Figure 16.

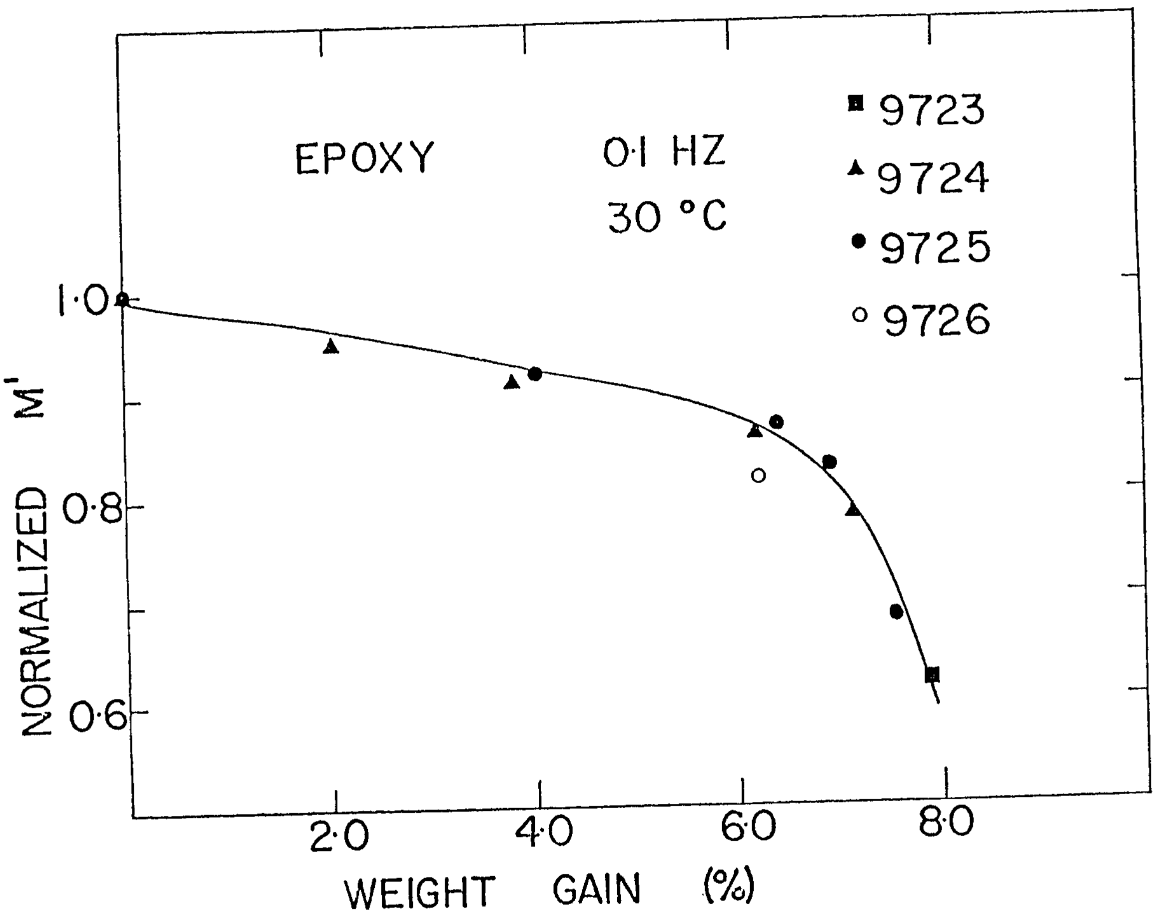


Figure 17.

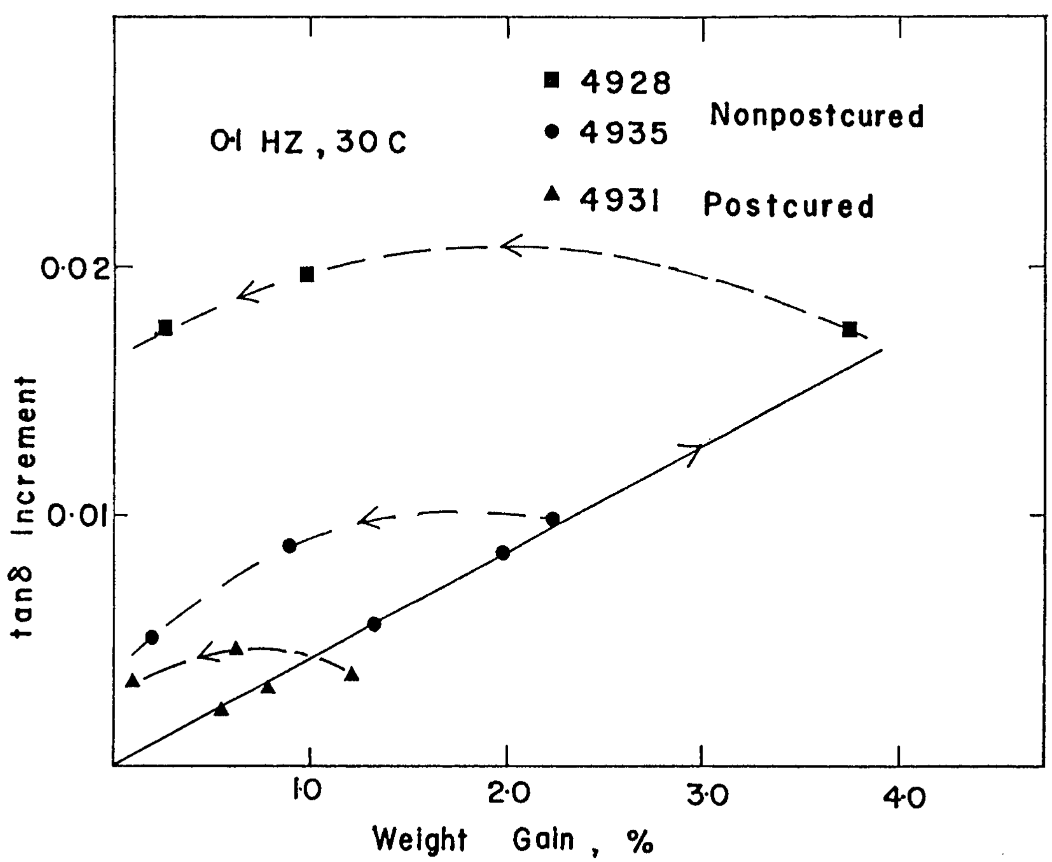


Figure 18.

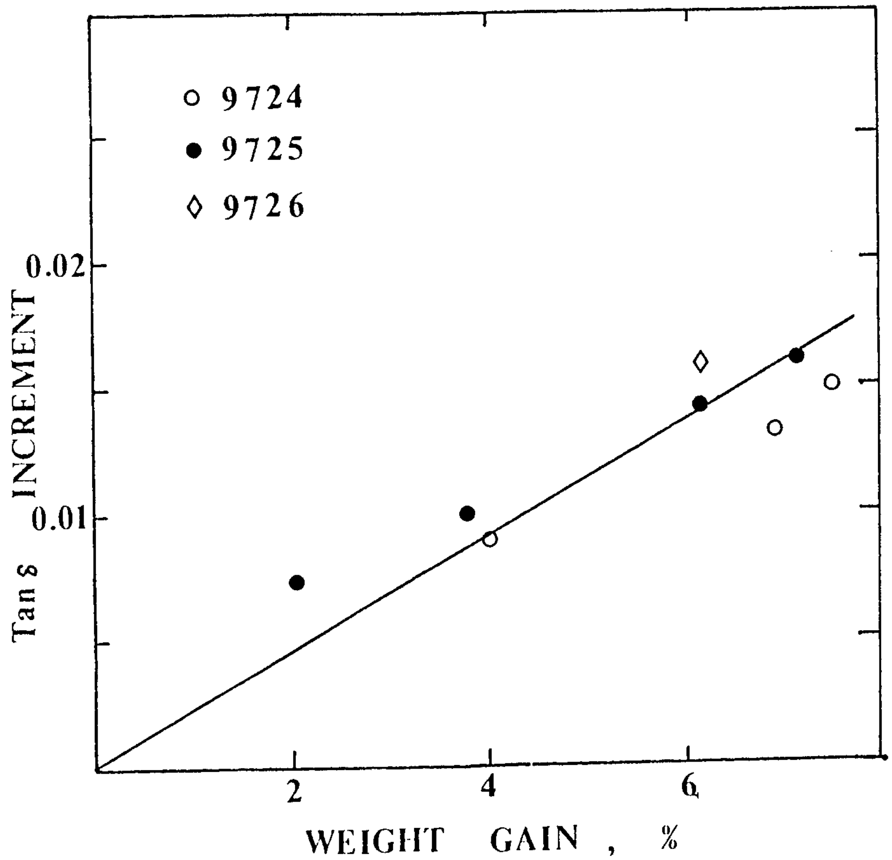
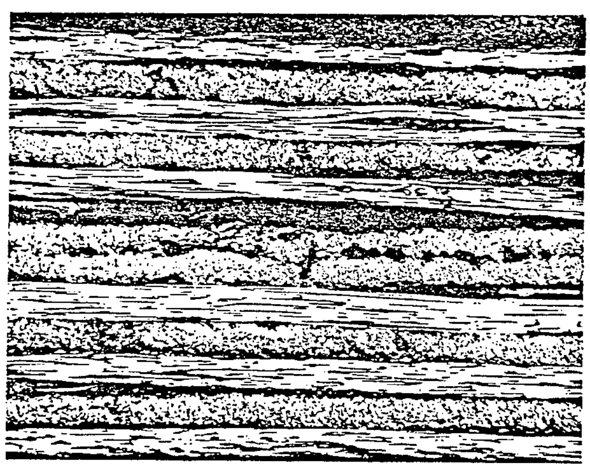


Figure 19.



WT. GAIN 5.1 %

50X

Figure 20.

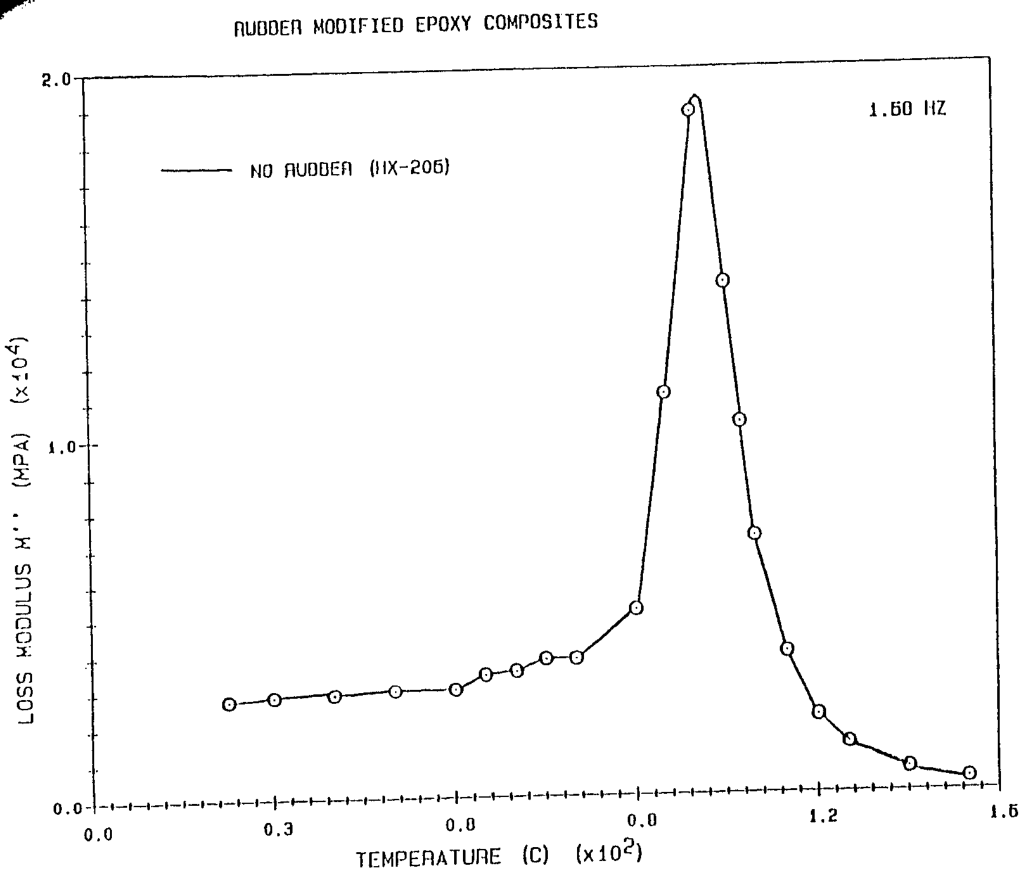


Figure 21.

BEAM, HXT #1, 32783

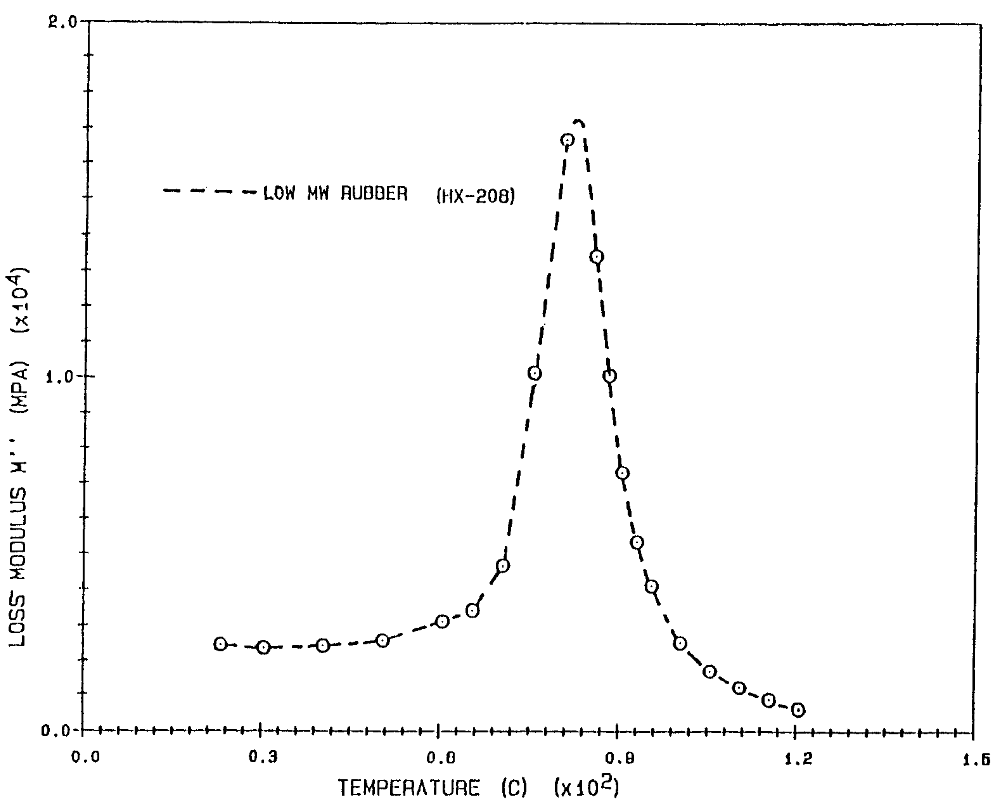


Figure 22.

1.58 Hz BEAM CHX #1, 32583

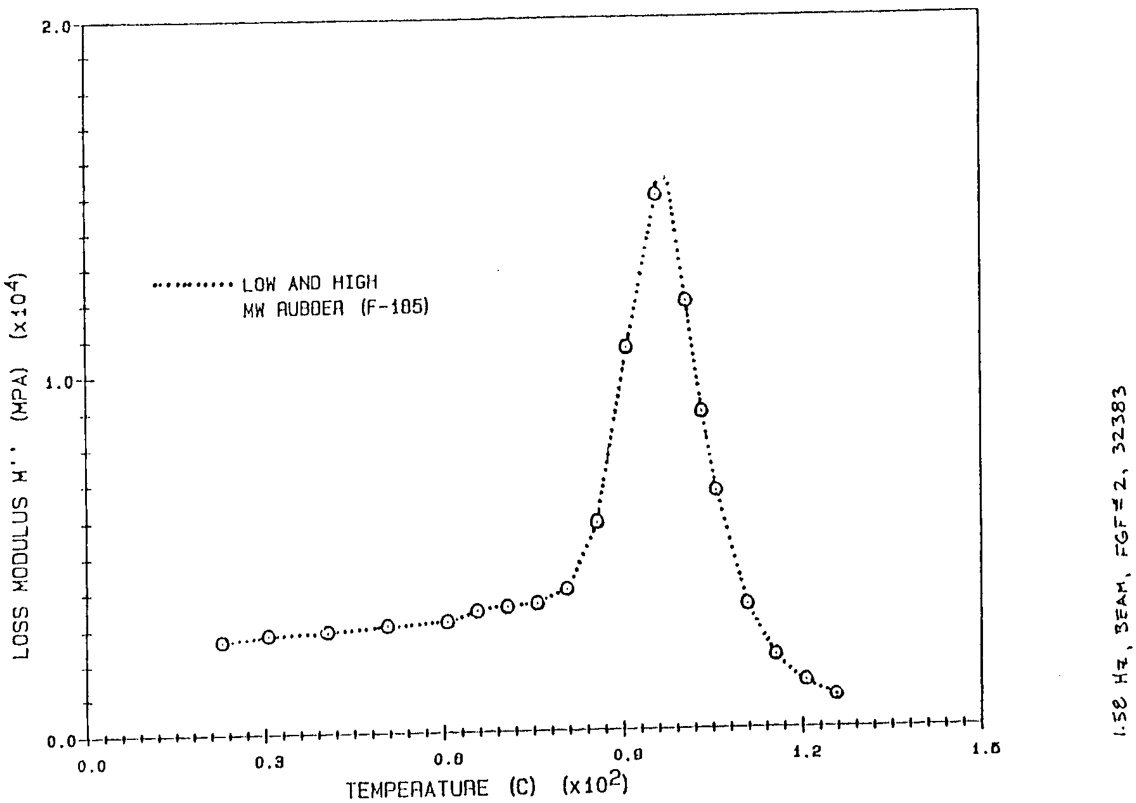


Figure 23.

NEAT POLYSULFONE

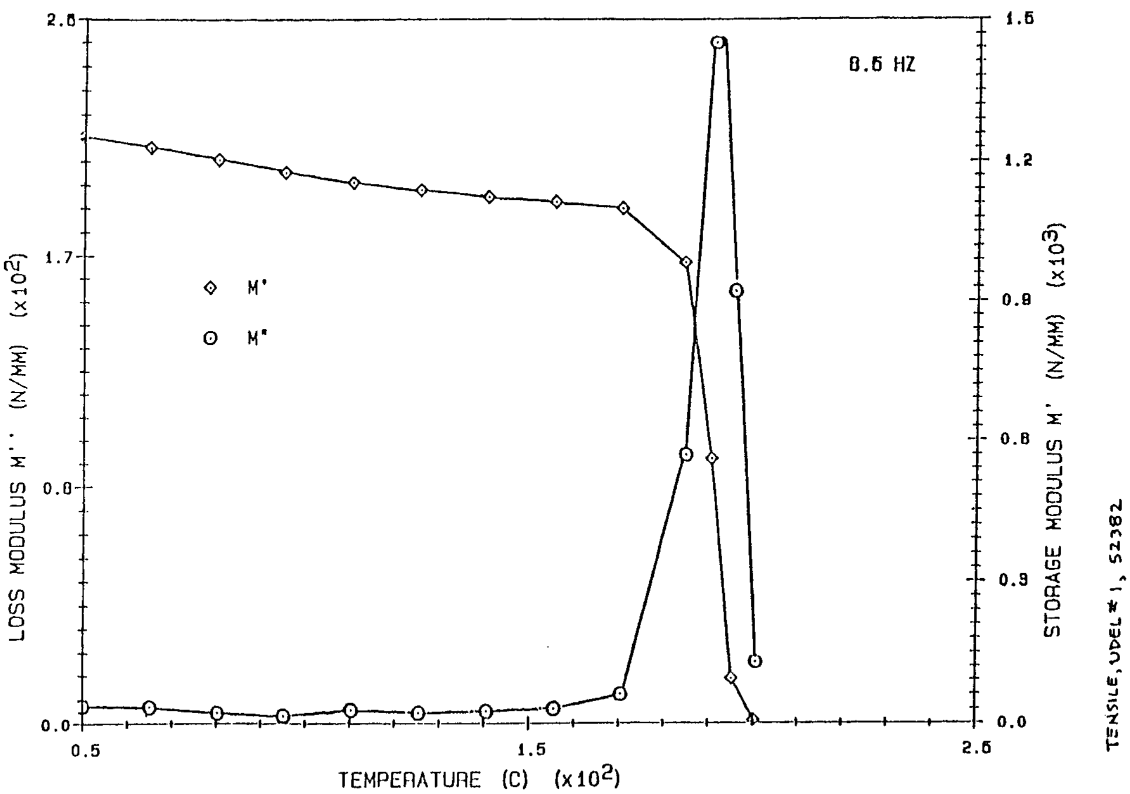
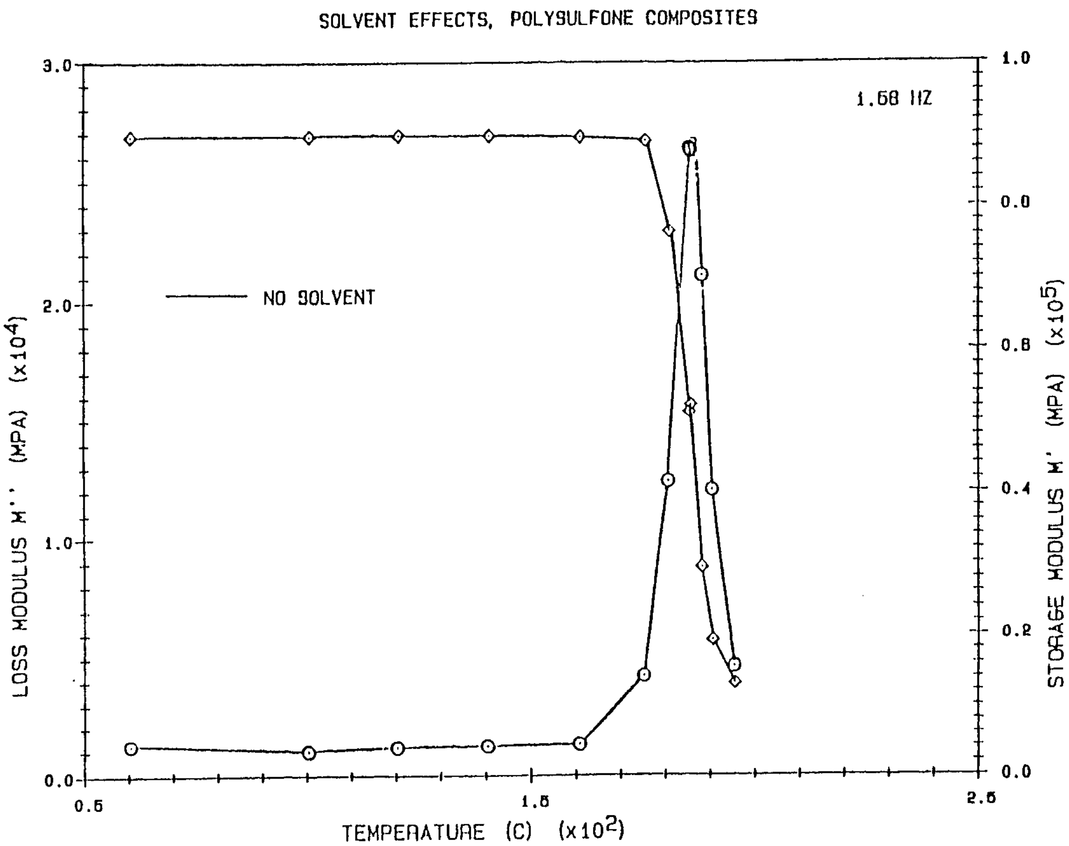


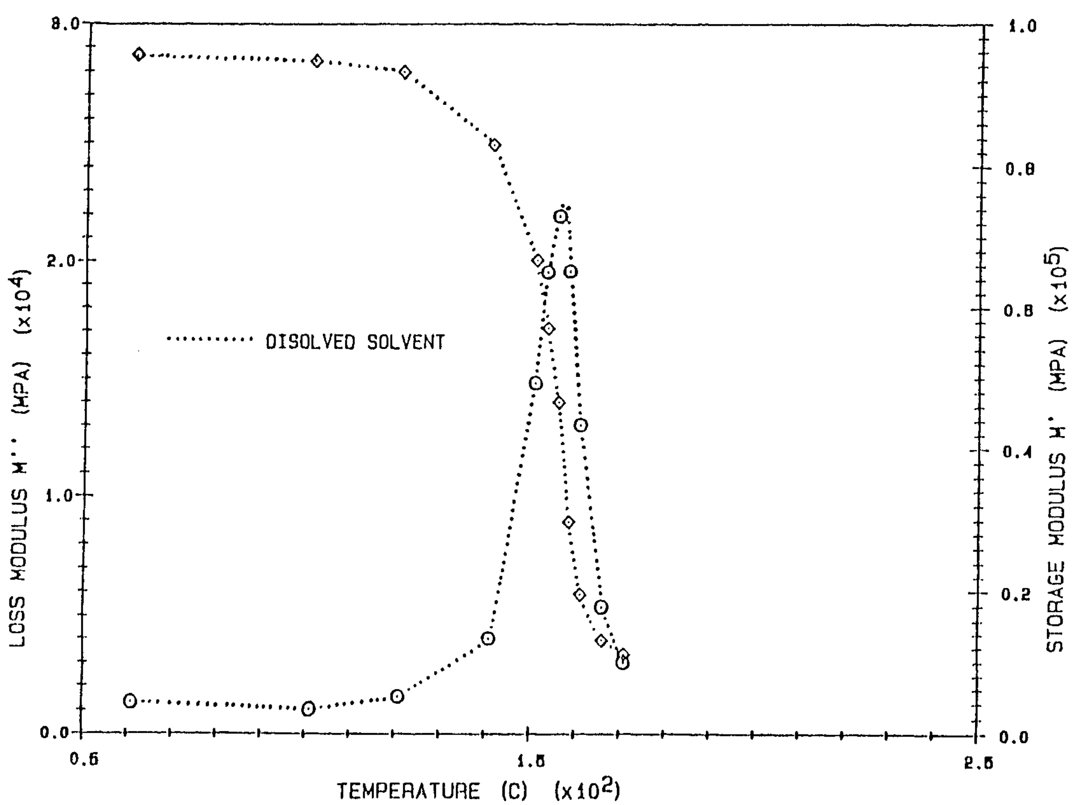
Figure 24.

Figure 25.



BEAM, CPS#1, 51883

Figure 26.



1.58 Hz, BEAM, GD-39 #6, 52083

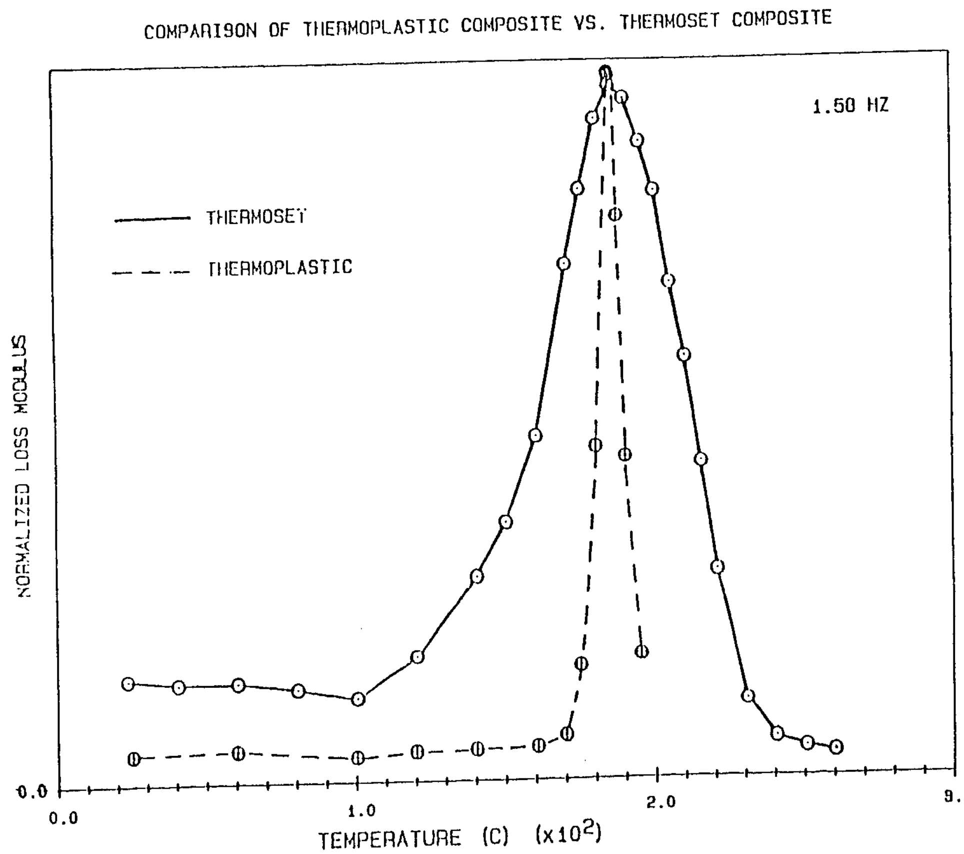


Figure 27.

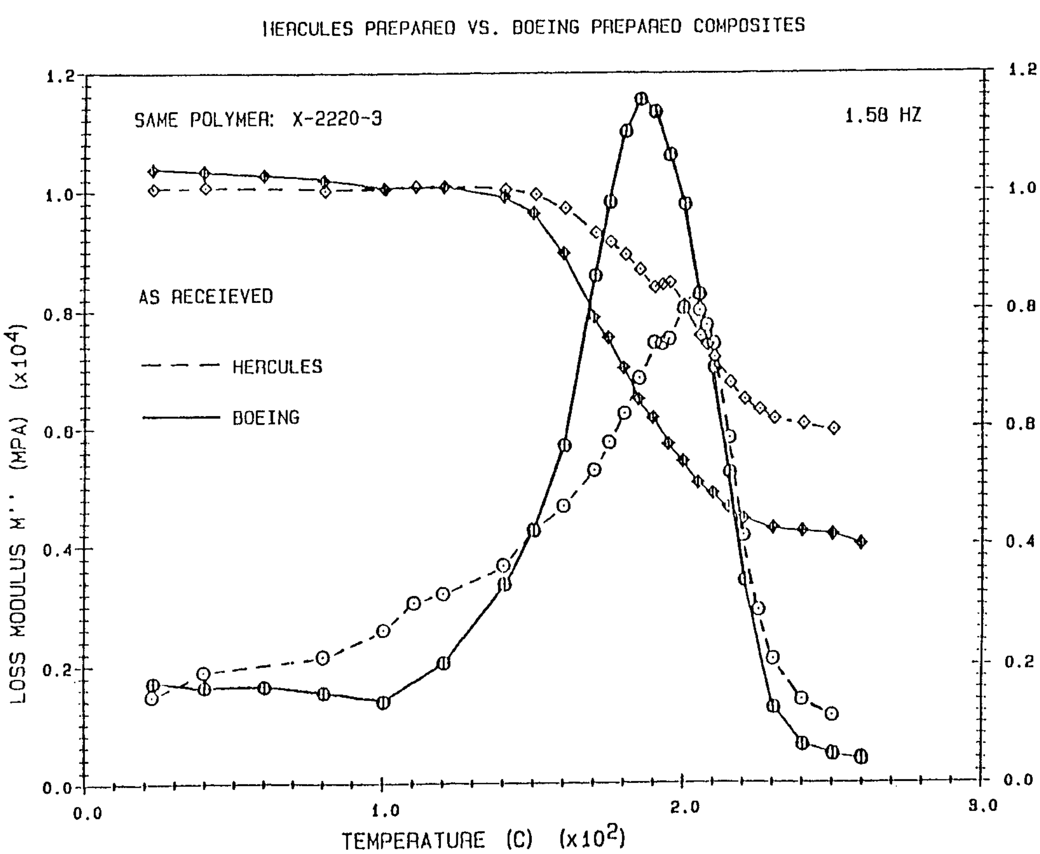


Figure 28.

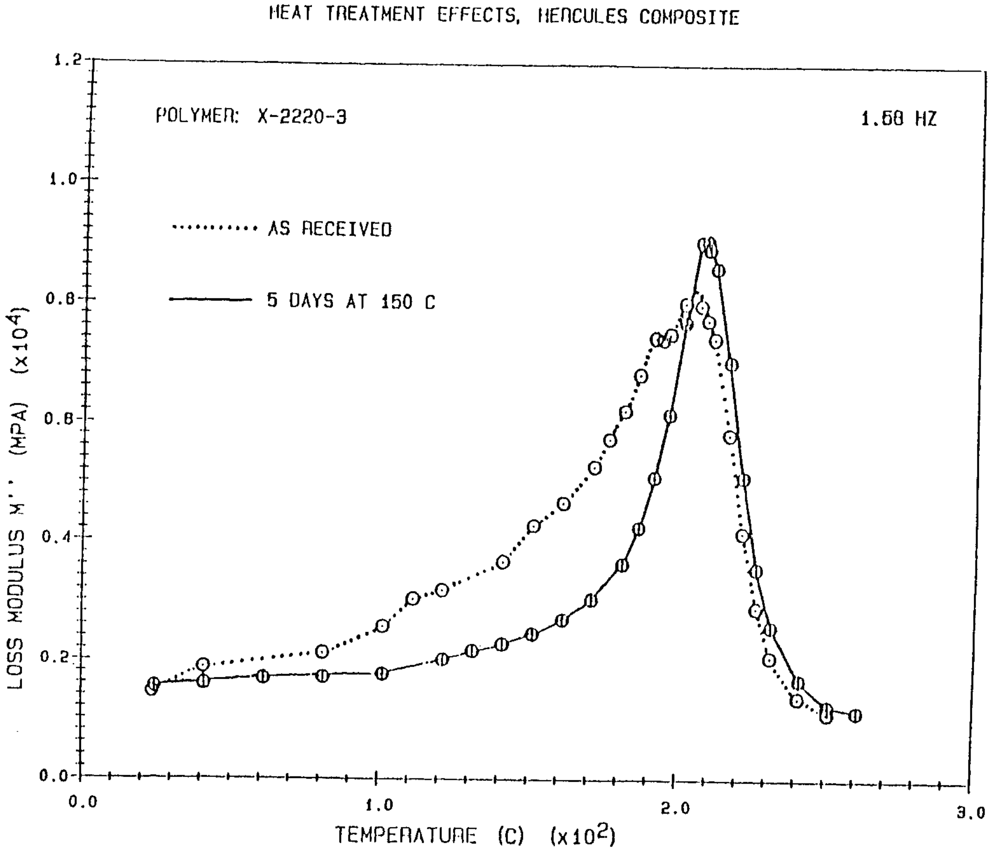


Figure 29.

BEAMS, XTT #6, 42583, XTT #5, 4583

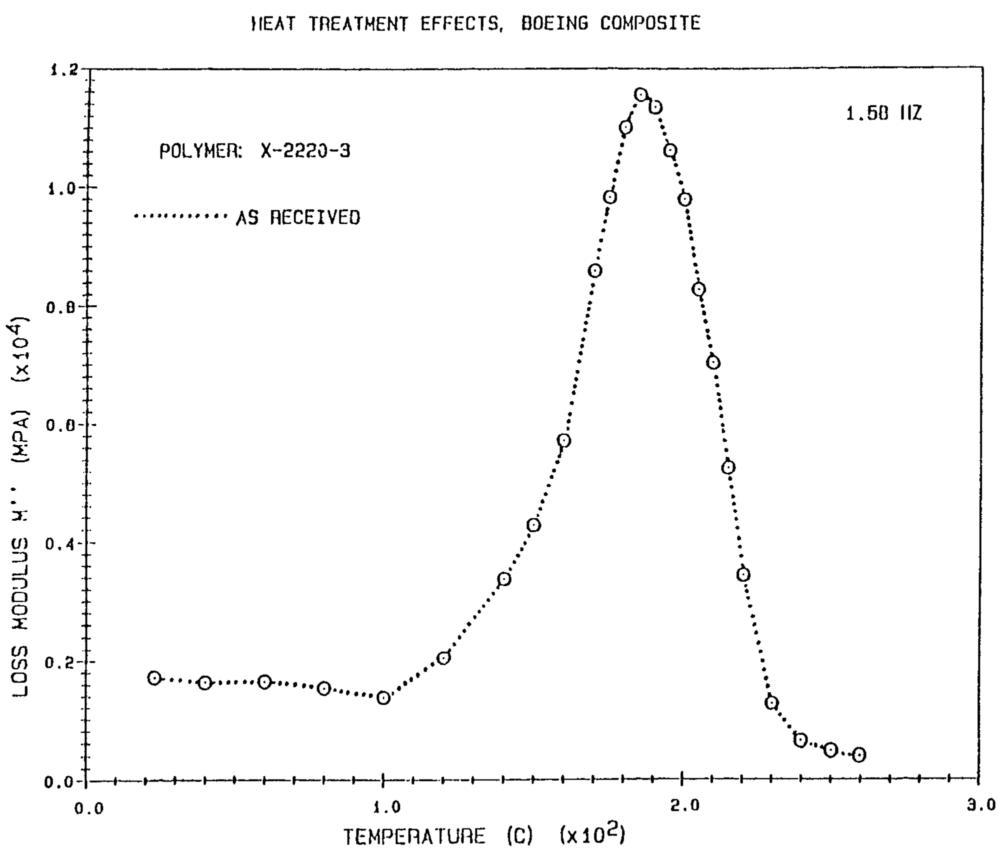


Figure 30.

BEAM, Boo #5, 31183

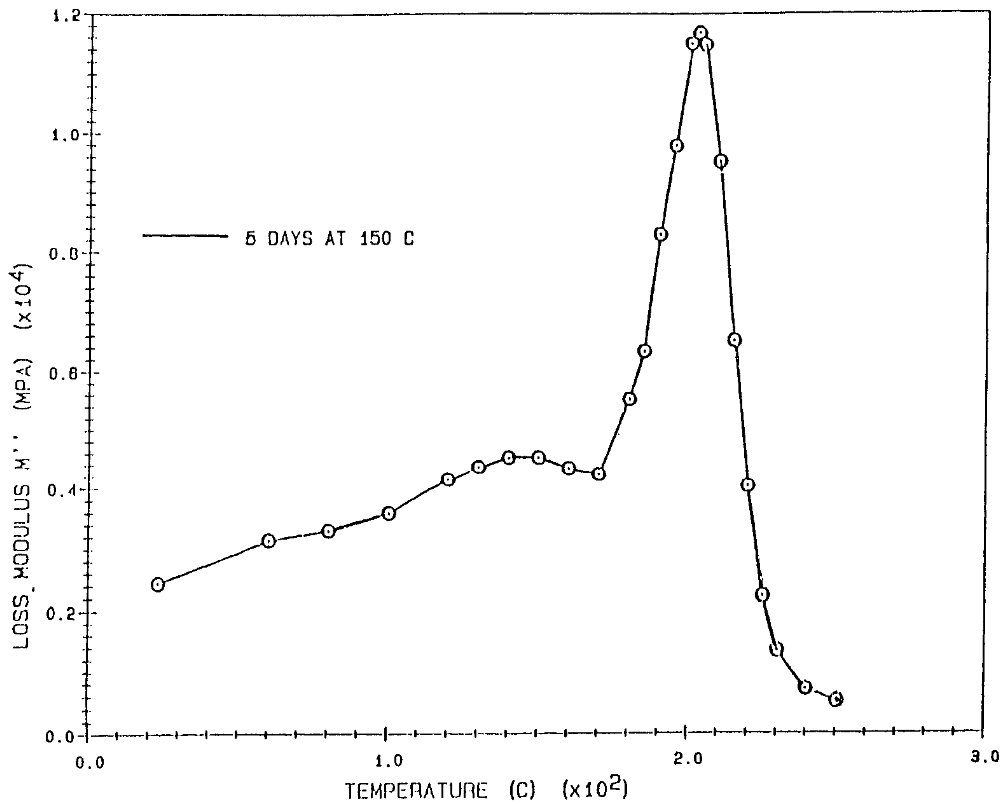


Figure 31.

1.58 Hz, BEAM, 300*7, 4683

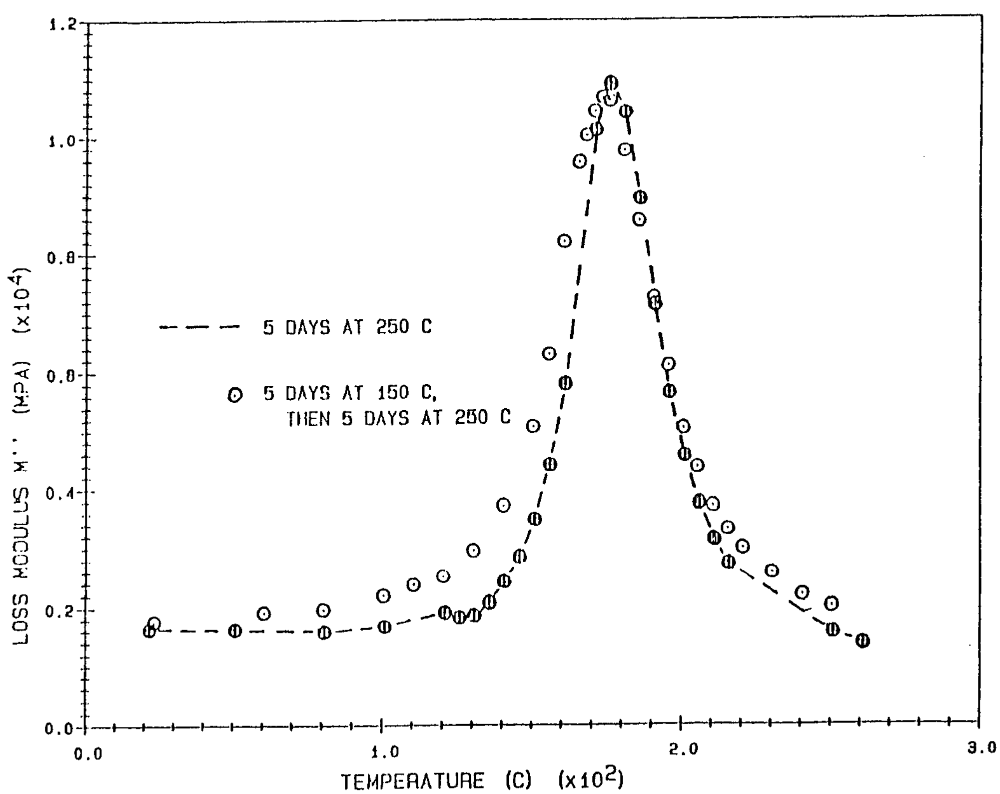


Figure 32.

1.58 Hz, BEAMS, 200*6, 31783, 300*7, 42683

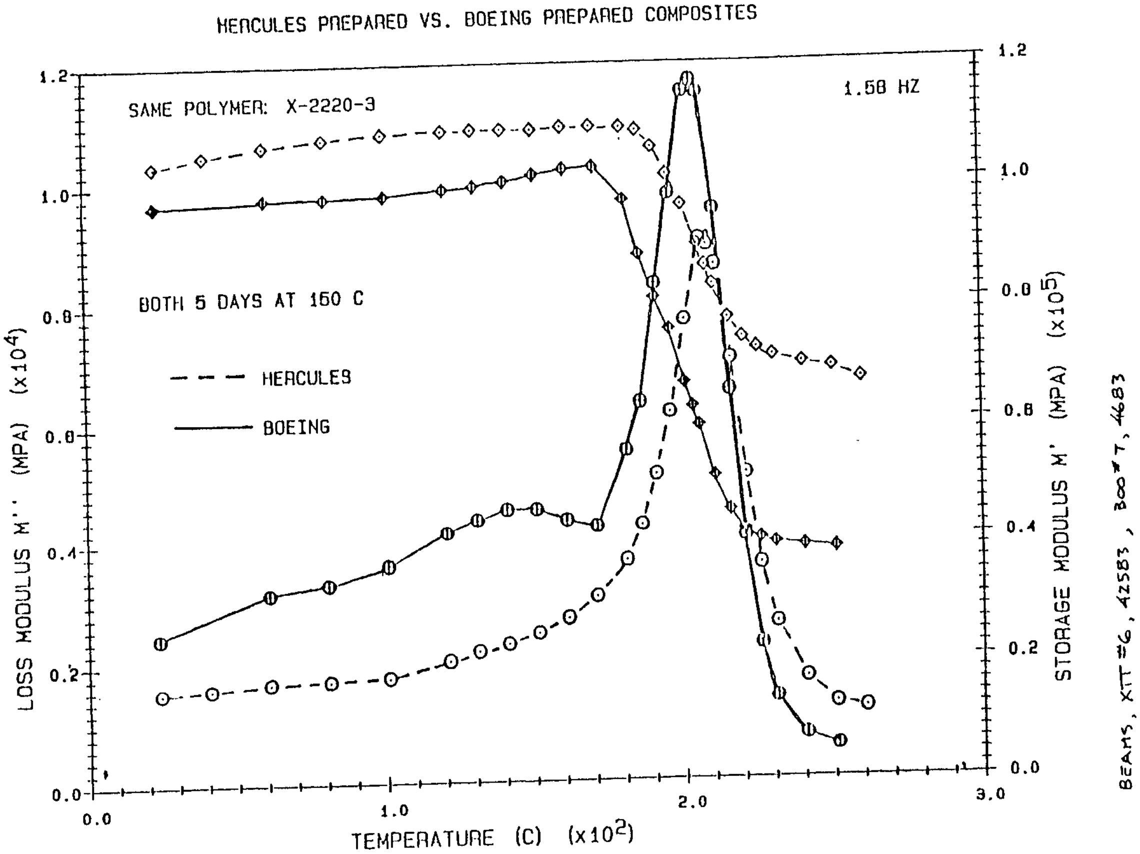


Figure 33.

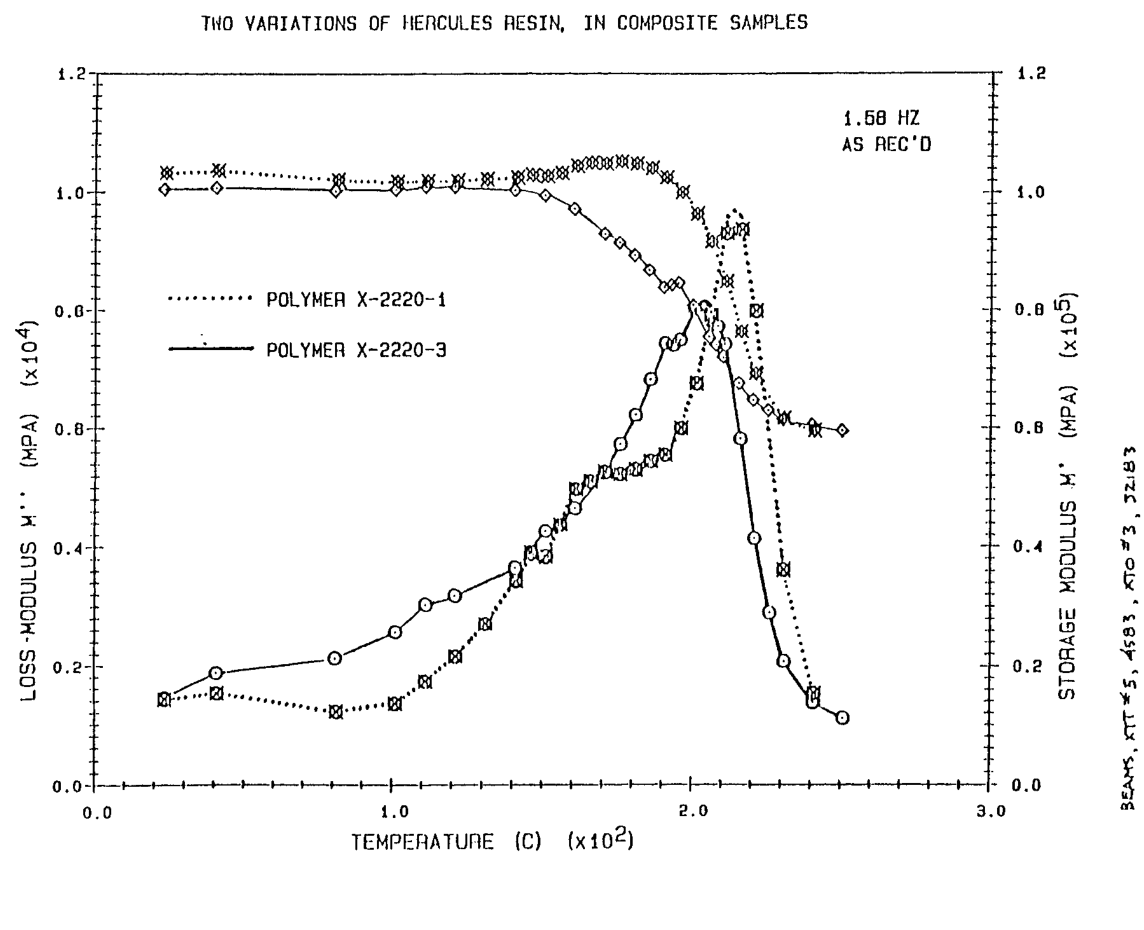


Figure 34.

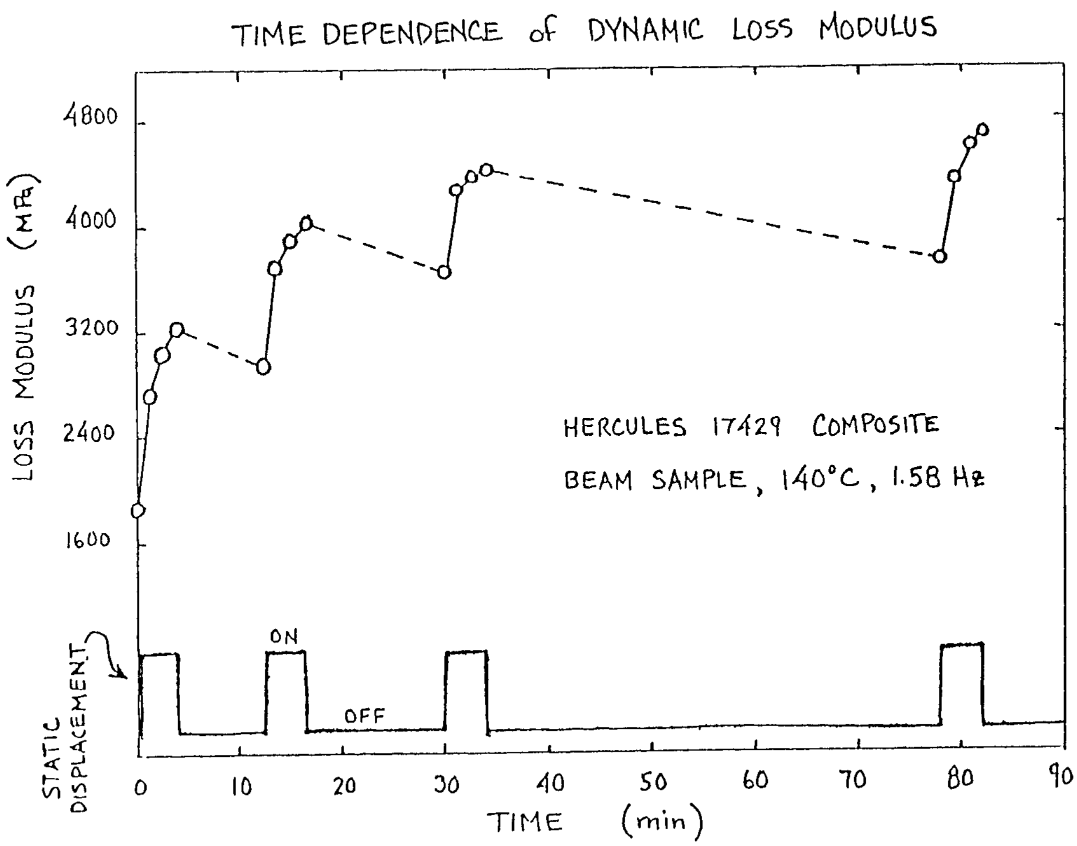


Figure 35.

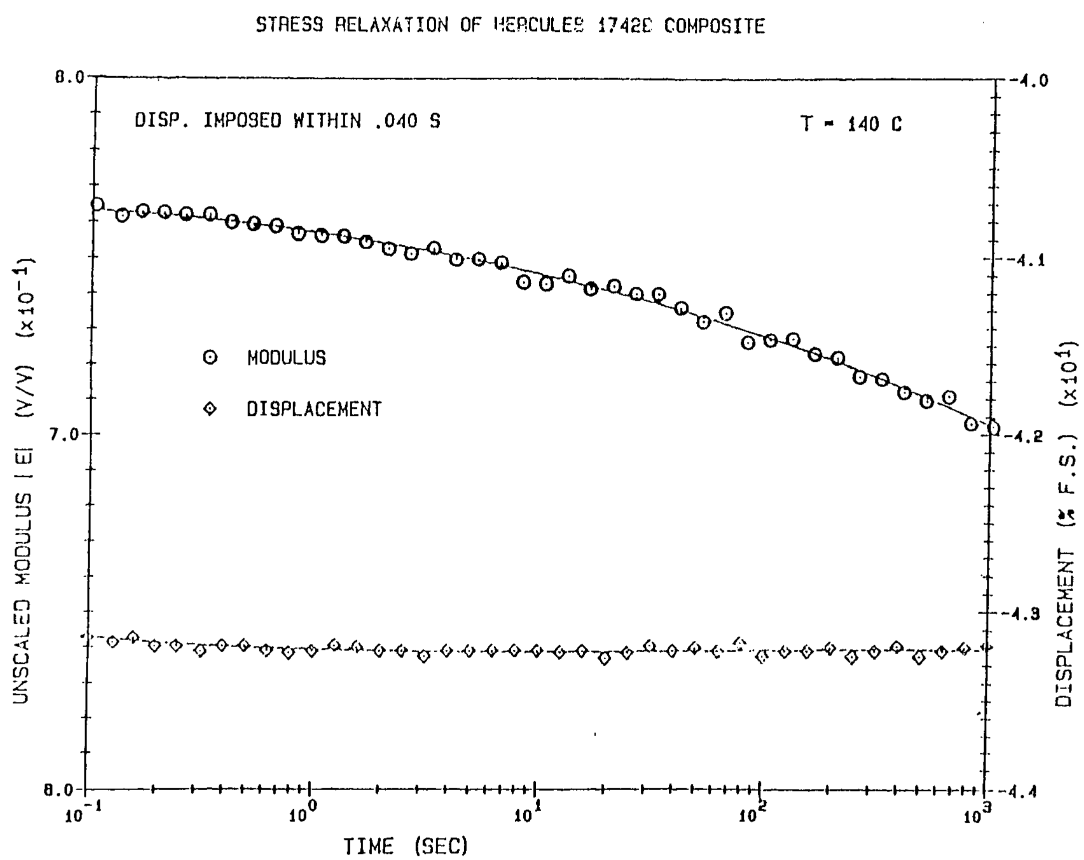


Figure 36.

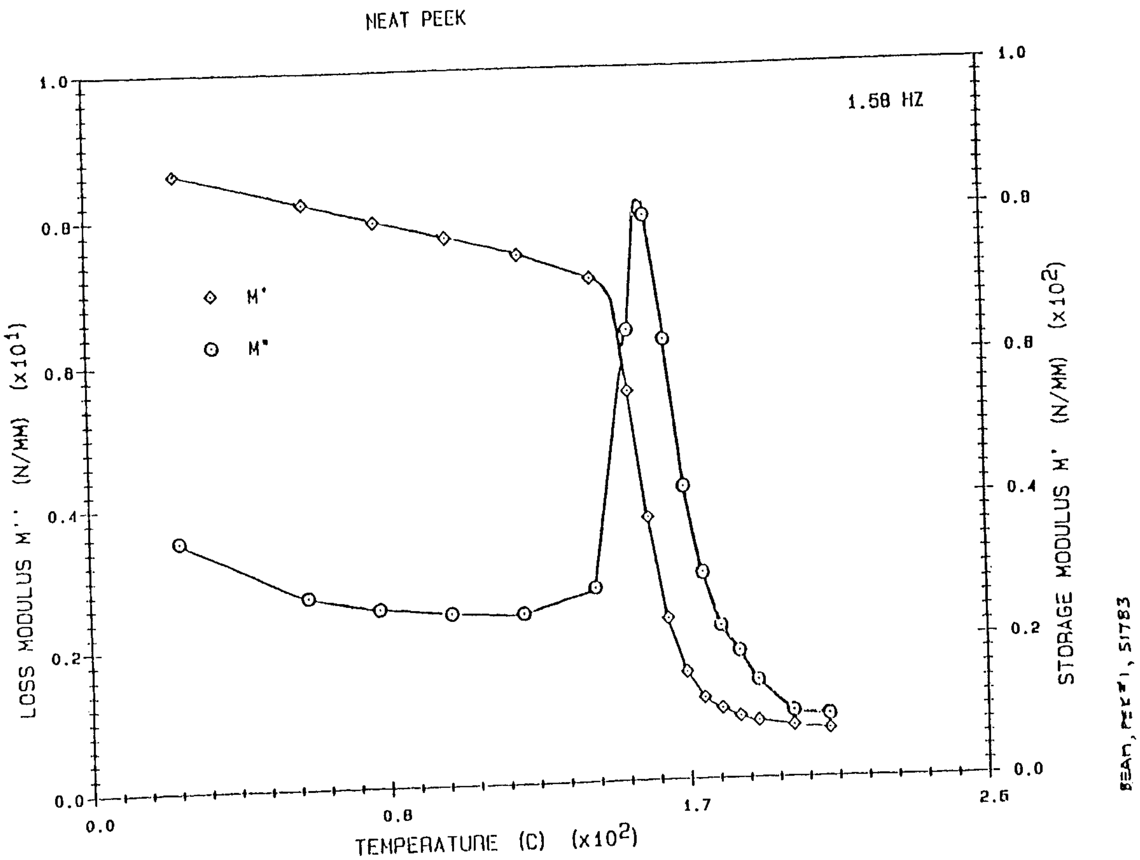


Figure 37.

Plys	Designation (increment)	Stacking Sequence
12	90°	$[(0/90)_3]_s$
12	60°	$[(0/60/120)_2]_s$
12	45°	$[(0/45/90/135/0/45)]_s$
12	30°	$[(0/30/60/90/120/150)]_s$
13	90°	$[(0/90)_3\,\overline{0}]_s$
13	60°	$[(0/60/120)_2\,\overline{0}]_s$
13	45°	$[(0/45/90/135/0/45)\,\overline{90}]_s$
13	30°	$[(0/30/60/90/120/150)\,\overline{0}]_s$

Stacking sequence designations for 12 and 13 ply laminates

Figure 38.

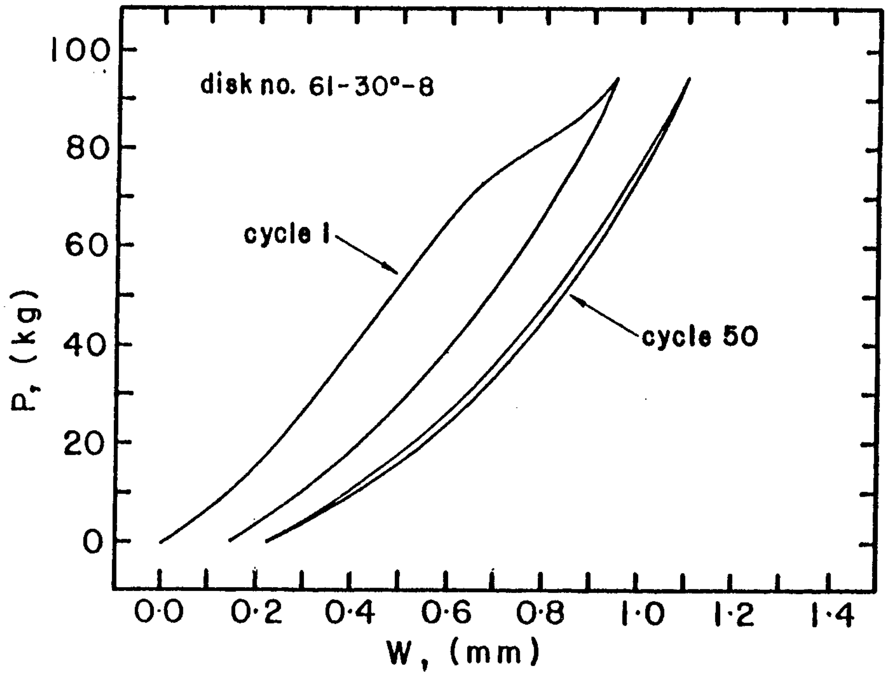


Figure 39. Load-deflection curve typical of 12ply and 13ply, 30°increment laminate disks.

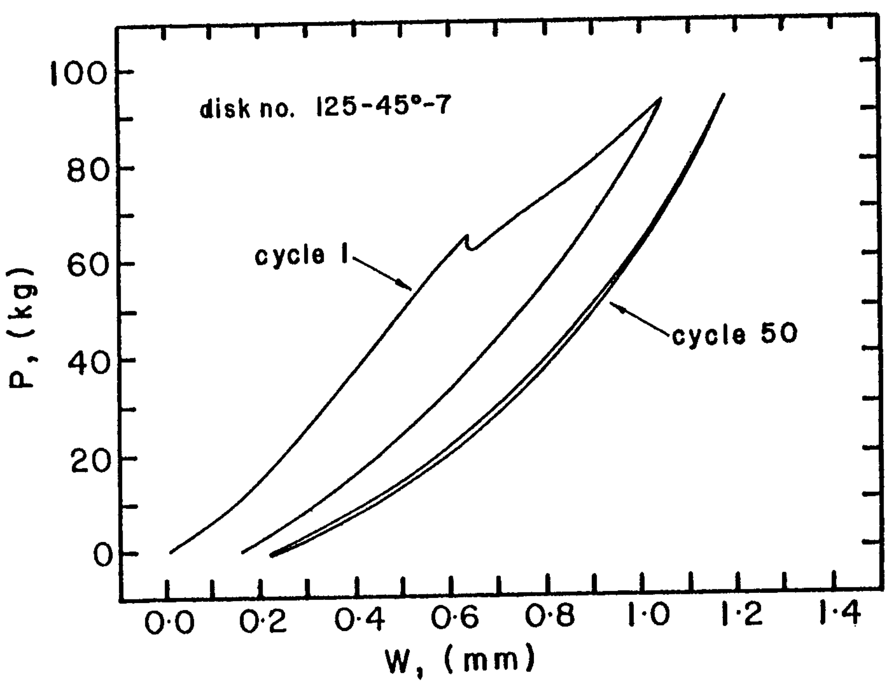
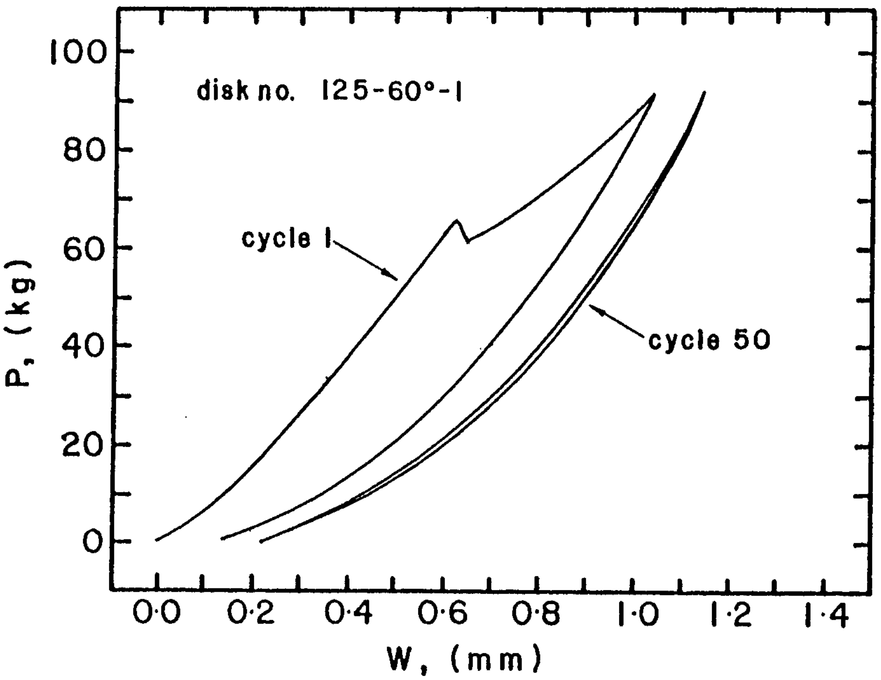


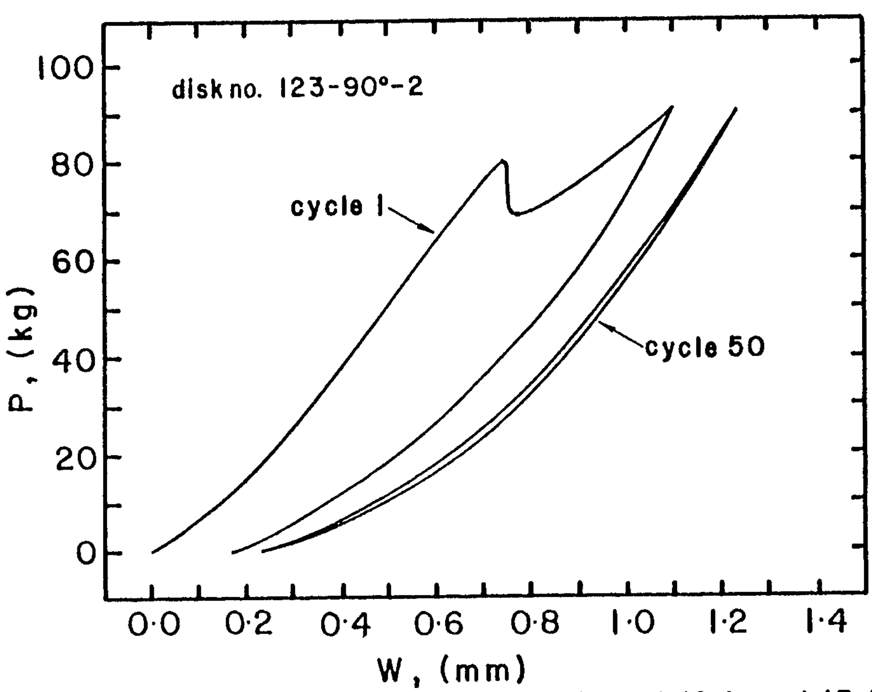
Figure 40. Load-deflection curve typical of 12ply and 13ply, 45° increment laminate disks.

Figure 41.

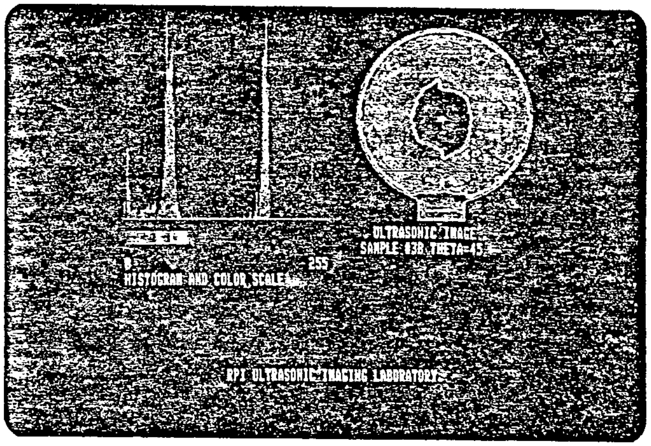


Load-deflection curve typical of 12ply and 13ply 60° increment laminate disks.

Figure 42.



Load-deflection curve typical of 12ply and 13ply 90° increment laminate disks.



P_{max}=95.0Kg; P_{br}=65.0Kg; ΔP_{br}= 3.0Kg.

Single cycle

C-scan of disk 26-45°-20

Figure 43.

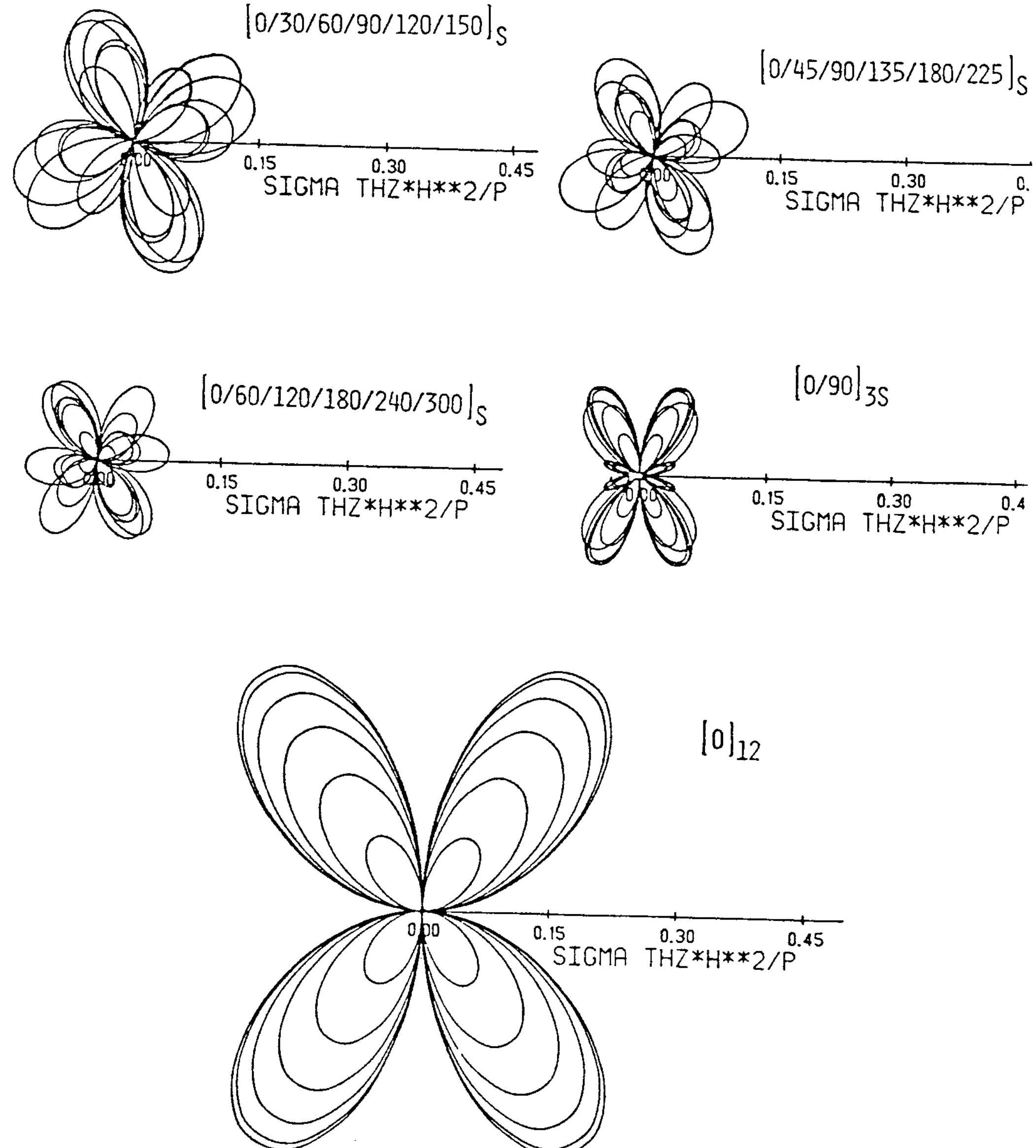


Figure 44.

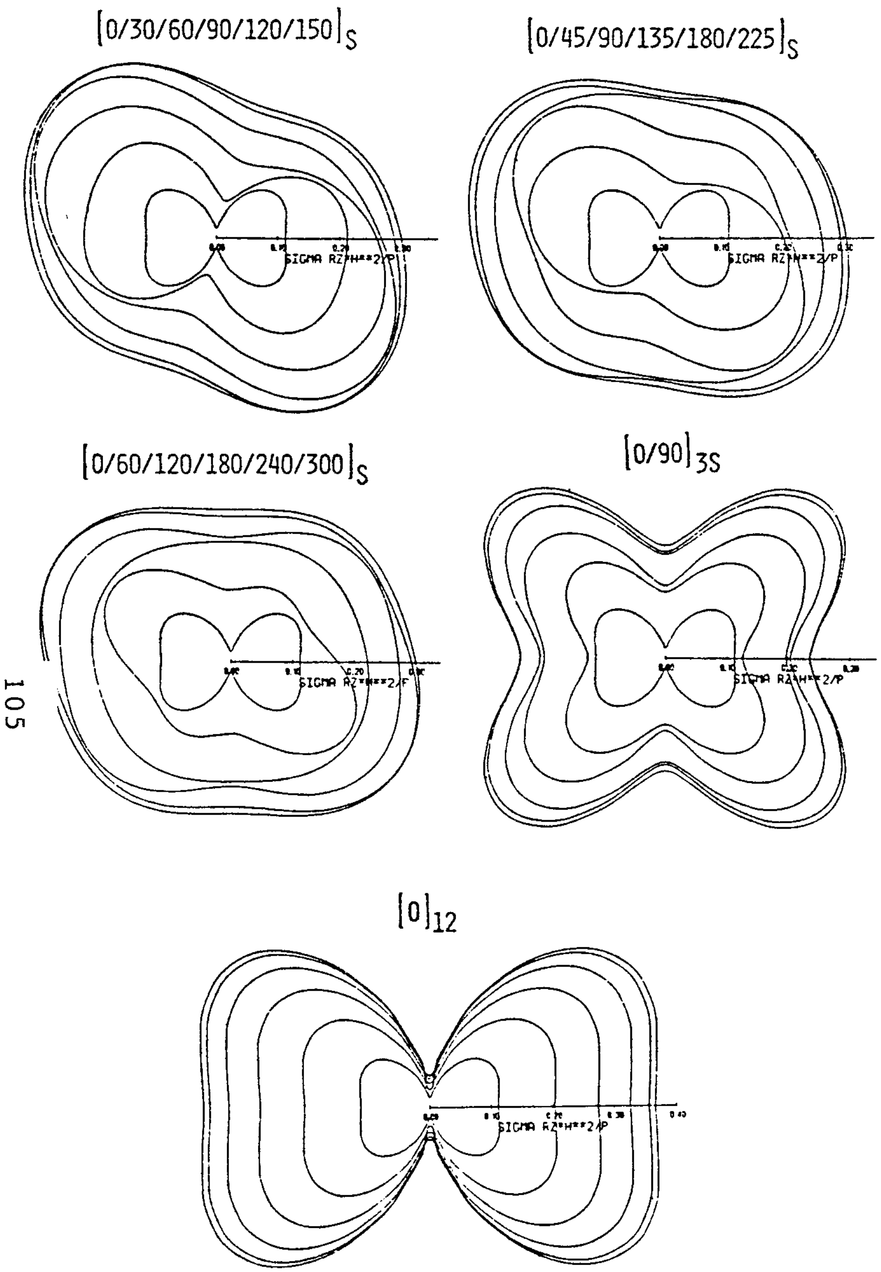

[0/30/60/90/120/150]$_S$
SIGMA RZ*H**2/P
[0/45/90/135/180/225]$_S$
SIGMA RZ*H**2/P
[0/60/120/180/240/300]$_S$
SIGMA RZ*H**2/P
[0/90]$_{3S}$
SIGMA RZ*H**2/P
[0]$_{12}$
SIGMA RZ*H**2/P

Figure 45

M.S. THESES

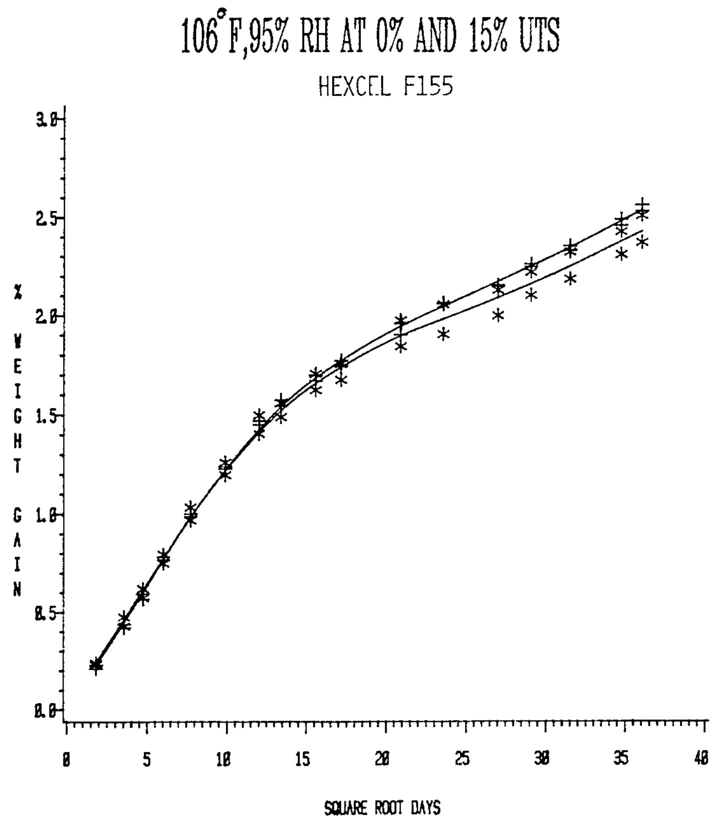
STUDENT/ FACULTY ADVISOR TITLE	OBJECTIVES
1. L. CLARK/ Y. WEITSMAN "MOISTURE DIFFUSION IN HYBRIDS"	COLLECT WEIGHT-GAIN DATA VS. TIME FOR F155 AND F185 GR/EP HYBRIDS. DEVELOP ANALYTICAL SOLUTION FOR DIFFUSION PROCESS EMPLOYING THE EXTENDED INNER PRODUCT CONCEPT AND JUMP CONDITION AT INTERFACE. COMPARE DATA AND MODEL PREDICTIONS.
2. E. PORTH/ Y. WEITSMAN "STRESS EFFECTS ON MOISTURE DIFFUSION"	COLLECT WEIGHT-GAIN DATA VS. TIME ON UNIDIRECTIONAL F155 COUPONS LOADED TRANSVERSLY AT SEVERAL STRESS LEVELS. DEVELOP THE COUPLED MOISTURE-DEFORMATION DIFFUSION EQUATION, INVESTIGATE SOURCES OF COUPLING AND COMPARE WITH EXPERIMENTAL RESULTS.
3. S. JACKSON/ Y. WEITSMAN "MOISTURE-INDUCED DAMAGE IN COMPOSITES"	EMPLOY SEM TO DETECT AND CHARACTERIZE DAMAGE IN CROSS-PLY AS/3502 LAMINATES DUE TO MOISTURE. COMPARE DAMAGE FORM AND EXTENT IN UNCONDITIONED, SATURATED, SATURATED-AND-DRIED, AND HUMIDITY CYCLED SPECIMENS.

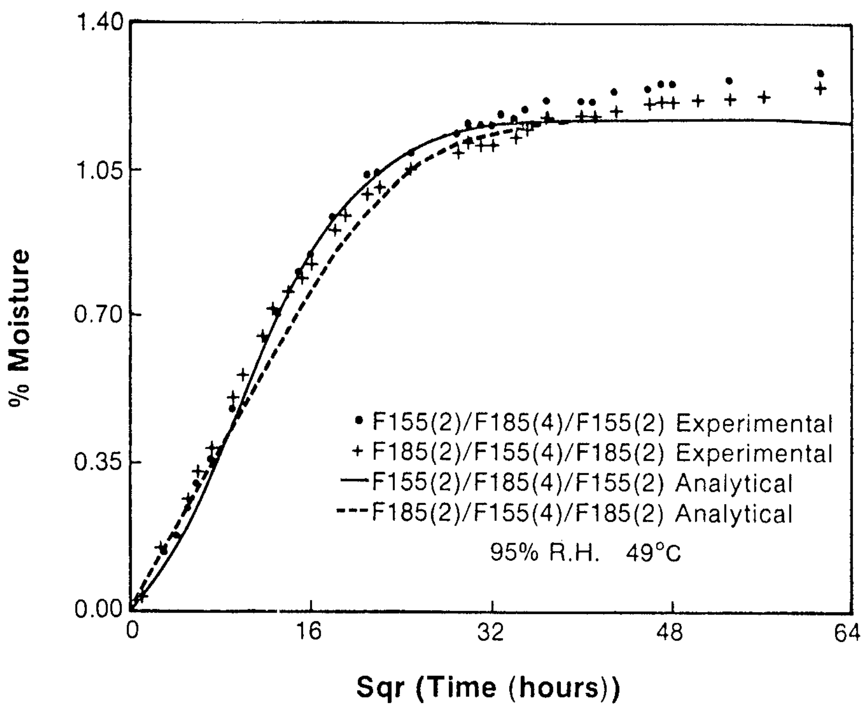
CONCLUSIONS

1) IF FICK'S LAW OF DIFFUSION IS OBEYED BY THE INDIVIDUAL COMPONENTS IT IS LIKELY TO APPLY ALSO TO THE HYBRID. THE PRECISE FORM OF THE INTERFACIAL BOUNDARY CONDITIONS REQUIRES FURTHER STUDY.

2) STRESS AFFECTS THE MOISTURE DIFFUSION PROCESS BY RAISING THE MOISTURE EQUILIBRIUM LEVEL AND BY A LOCAL INTERACTION BETWEEN VOLUMETRIC STRAINS AND MOISTURE TRANSPORT. THE LATTER EFFECTS CANNOT BE SINGLED OUT WHEN THE STRESS-INDUCED DEFORMATION IS SPATIALLY HOMOGENEOUS.

3) DRYING AND/OR CYCLIC RELATIVE HUMIDITY ARE ENVIRONMENTS WHICH ARE MORE SEVERE THAN A FULLY SATURATED CASE.





Moisture Absorption in Hybrid Graphite-Epoxy Reinforced Composites

ON THE EFFECTS OF POST CURE COOL DOWN AND ENVIRONMENTAL CONDITIONING ON RESIDUAL STRESSES IN COMPOSITE LAMINATES

PH.D. THESIS
BY
B.D. HARPER

OBJECTIVES

1) DEVELOP A MATHEMATICAL MODEL AND EXPERIMENTAL CHARACTERIZATION METHOD FOR THERMORHEOLOGICALLY COMPLEX VISCOELASTIC MATERIALS AND APPLY TO HERCULES 3502 EPOXY RESIN.

2) ASSESS THE EFFECTS OF CHEMICAL CURE SHRINKAGE STRAINS UPON RESIDUAL STRESSES.

3) PREDICT OPTIMAL COOL DOWN PATHS WHICH MINIMIZE RESIDUAL THERMAL STRESSES IN BALANCED, SYMMETRIC CROSS-PLY LAMINATES.

4) STUDY THE APPLICABILITY OF LINEAR VISCOELASTICITY IN PREDICTING CURVATURES OF ANTI-SYMMETRIC CROSS-PLY GRAPHITE/EPOXY LAMINATES AFTER BEING COOLED FROM THEIR CURE TEMPERATURE AND DURING EXPOSURE TO BOTH CONSTANT AND FLUCTUATING RELATIVE HUMIDITIES. BOTH ELASTIC AND VISCOELASTIC ANALYTICAL PREDICTIONS ARE COMPARED WITH MEASURED CURVATURES OF AS4/3502 GRAPHITE/EPOXY LAMINATES.

CONCLUSIONS

1) A CORRECT TIME-TEMPERATURE ANALOGY SHOULD INCLUDE VERTICAL AS WELL AS HORIZONTAL SHIFT FACTORS. APPROPRIATE TRANSIENT TEMPERATURE CHARACTERIZATION TESTS ARE REQUIRED TO UNIQUELY DETERMINE THE HORIZONTAL AND VERTICAL SHIFT FACTORS. THIS IS IMPORTANT BECAUSE LONG-TIME BEHAVIOR CAN BE PREDICTED FROM SHORT-TIME HIGH TEMPERATURE TESTS ONLY THROUGH THE HORIZONTAL SHIFT.

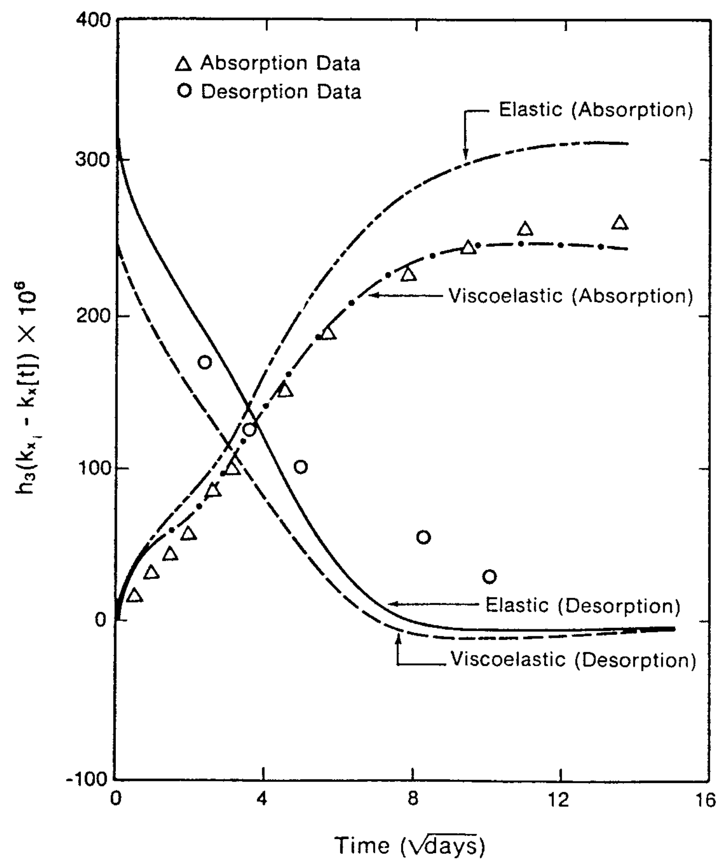
2) CHEMICAL CURE SHRINKAGE EFFECTS ARE SIGNIFICANT FOR RESINS CURED BELOW T_G. THESE EFFECTS ACCOUNTED FOR APPROXIMATELY 30% OF THE RESIDUAL STRESS IN A LAMINATED HERCULES 3502 RESIN/ALUMINUM COUPON AT ROOM TEMPERATURE.

3) TEMPERATURE DEPENDENCE OF THE THERMAL EXPANSION COEFFICIENTS HAS LITTLE EFFECT UPON THE OPTIMAL COOL-DOWN PATH. HOWEVER, IT HAS A SIGNIFICANT EFFECT UPON THE ASSOCIATED RESIDUAL THERMAL STRESSES.

4) CURVATURE MEASUREMENTS OF ANTI-SYMMETRIC CROSS-PLY AS4/3502 GRAPHITE/EPOXY LAMINATES INDICATE THAT VISCOELASTIC EFFECTS ARE OF SECONDARY IMPORTANCE DURING COOL DOWN, BUT BECOME SIGNIFICANT IN THE PRESENCE OF MOISTURE. BOTH CURVATURE MEASUREMENTS AND WEIGHT GAIN DATA INDICATE THE PRESENCE OF IRREVERSIBLE MOISTURE-INDUCED DEGRADATION OF MATERIAL PROPERTIES. THIS CONTENTION IS FURTHER SUPPORTED BY SEM STUDIES. THE MOST DETRIMENTAL EFFECT OF MOISTURE SEEMS TO BE ENCOUNTERED DURING DRYING, WHICH CAUSES MORE DAMAGE THAN UNDER FULLY SATURATED CONDITIONS.

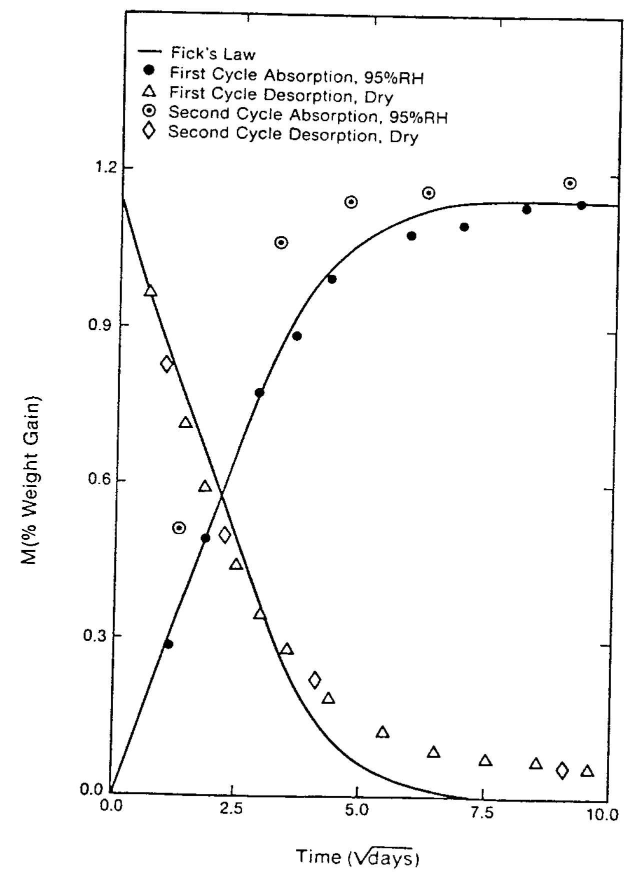
7

TIME-DEPENDENT CURVATURE CHANGE FOR ABSORPTION/DESORPTION AT 150°F, 75% RH

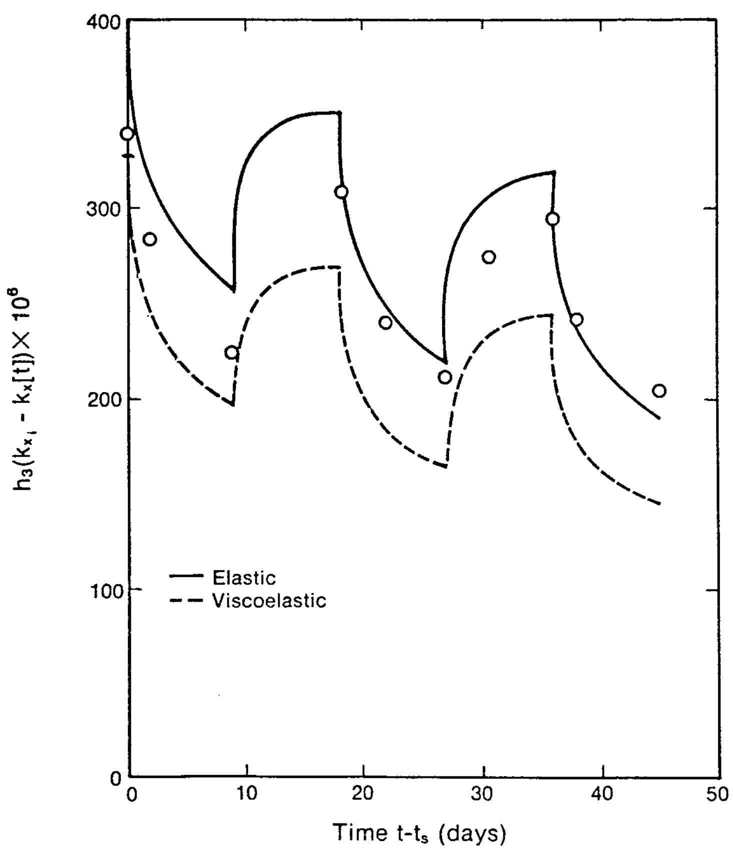


8

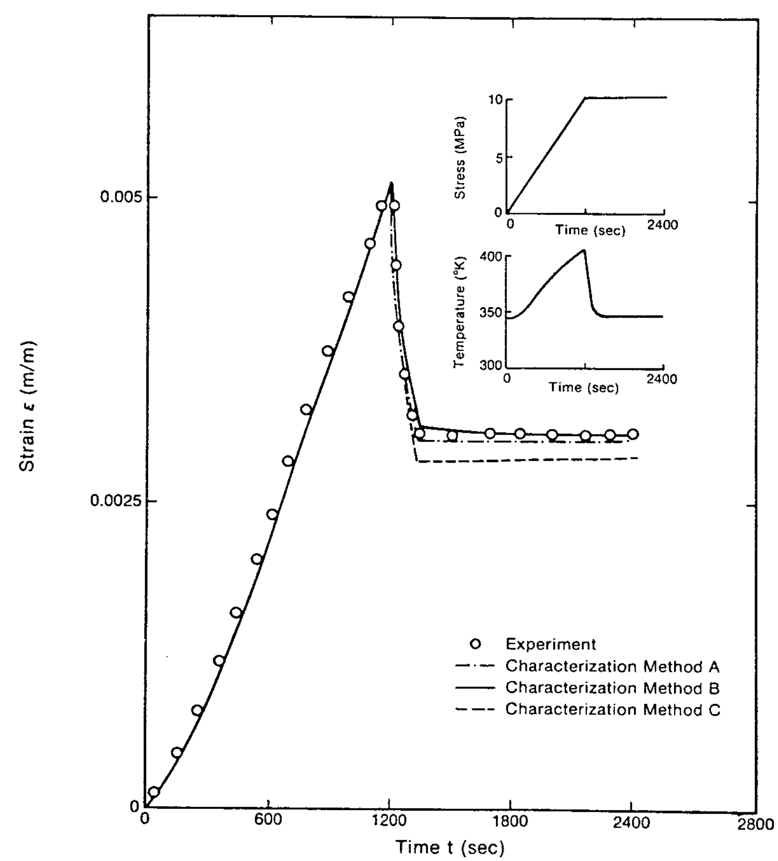
WEIGHT GAIN FOR ABSORPTION / DESORPTION AT 163°F, 95% RH



TIME–DEPENDENT CURVATURE CHANGE DURING CYCLIC EXPOSURE TO 0 AND 95% RH AT 130°F



THERMOVISCOELASTIC MODEL VERIFICATION



EFFECT OF STRAIN RATE ON GRAPHITE/EPOXY LAMINATES

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O B J E C T I V E

DETERMINE THE EFFECT OF STRAIN RATE

ON THE STRENGTH OF GRAPHITE/EPOXY LAMINATES

CONCLUSIONS

STRAIN RATE EFFECTS DO EXIST BUT ARE NOT CRITICAL TO THE EVALUATION OF LAMINATE STRENGTH OF AIRCRAFT–TYPE LAMINATES.

- 7 OUT OF 15 TEST CASES SHOW STATISTICALLY SIGNIFICANT RESULTS. 6 OF THE 7 CASES INDICATE THAT Gr/Ep LAMINATES ARE STRONGER AT FASTER STRAIN RATES. (IN GENERAL, STATIC TESTING IS CONDUCTED AT THE SLOWER STRAIN RATES).

- STRAIN RATE SENSITIVITY IS MORE APPARENT AT ETW THAN RTD. THE MEAN STRENGTHS UNDER ETW COMPRESSION SHOWED DIFFERENCES FROM 12% (IN AIRCRAFT–TYPE LAMINATES) TO 25% (IN THE 0 LAMINATE).

APPROACH

- SPECIMEN GEOMETRY
- LAMINATE LAYUP
  $[0]_{24T}$; $[\pm 45]_{12S}$; $[(\pm 45/0_2)_3/\overline{90}]_S$; $[(\pm 45/0/\pm 45)_2/\pm 45/\overline{90}]_S$
- ENVIRONMENT
  RTD; ETW (200 F, 1% MOISTURE)
- LOADING
  MONOTONIC TENSION AND COMPRESSION
- LOADING RATE
  FAILURE IN 0.1, 1.0, 10, AND 100 SECONDS
- REPLICATES
  RTD: 7–10 SPECIMENS PER CONDITION
  ETW: 9–11 SPECIMENS PER CONDITION

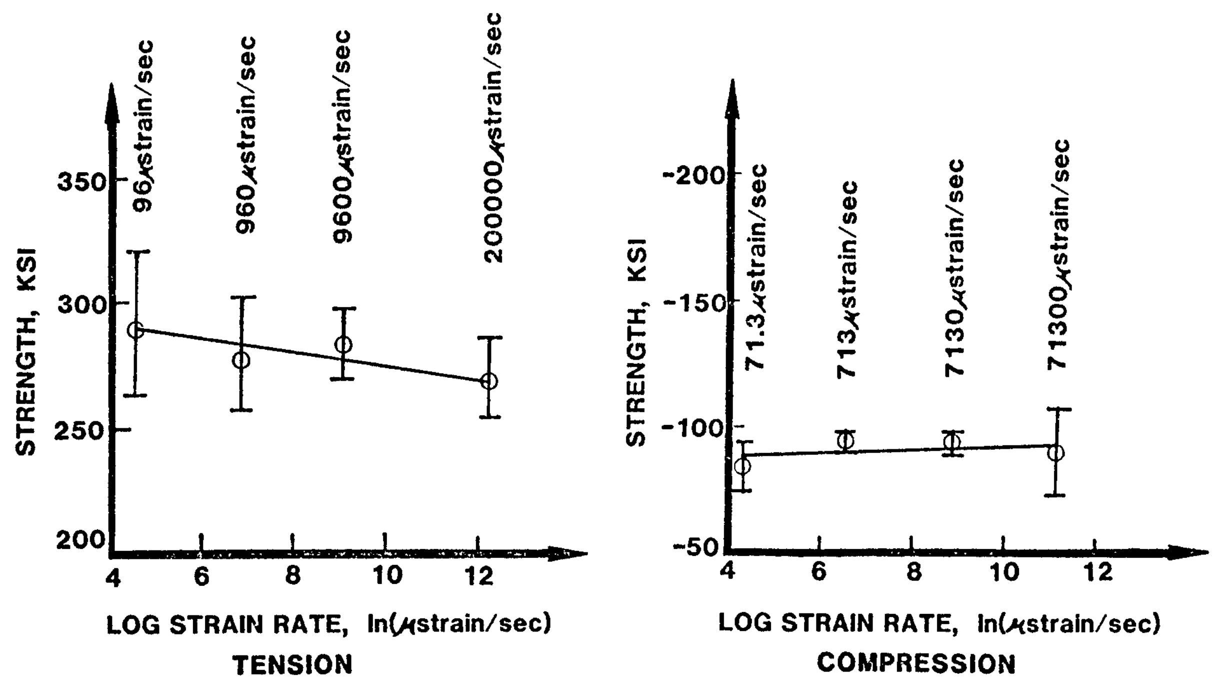


Figure 1 MEAN STRENGTH±STANDARD DEVIATION

LAMINATE 1 (100/0/0) RTD

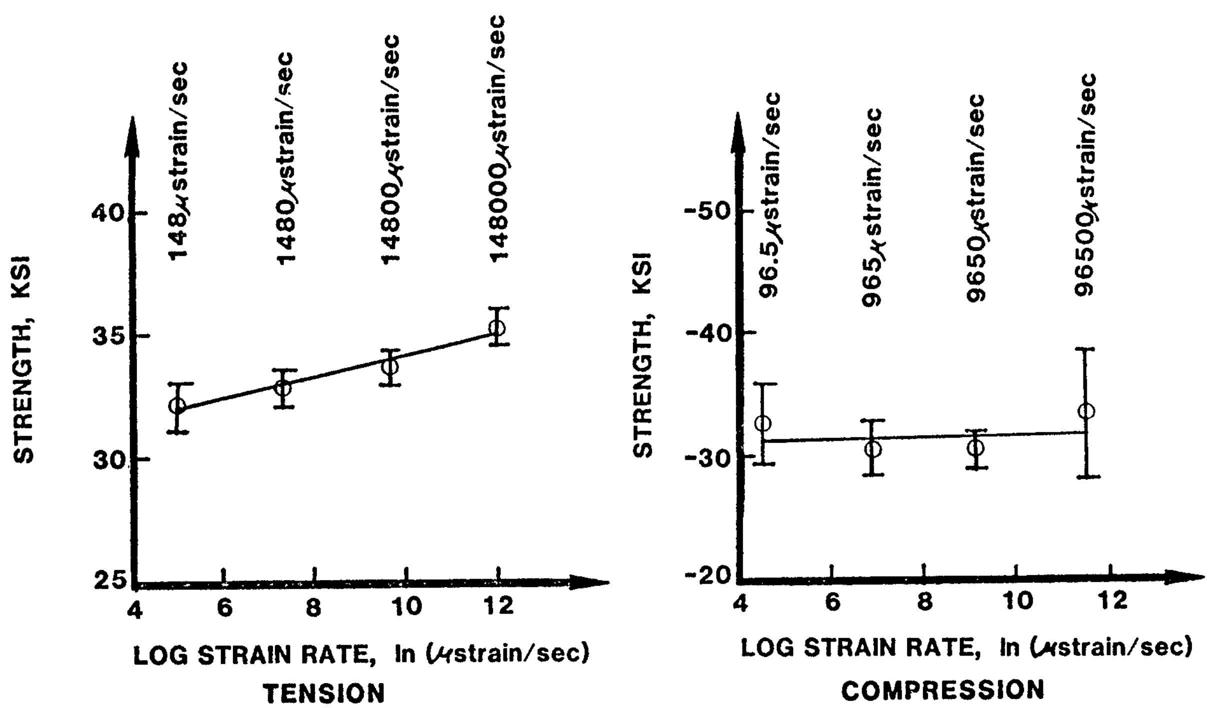


Figure 2 MEAN STRENGTH±STANDARD DEVIATION

LAMINATE 2 (0/100/0) RTD

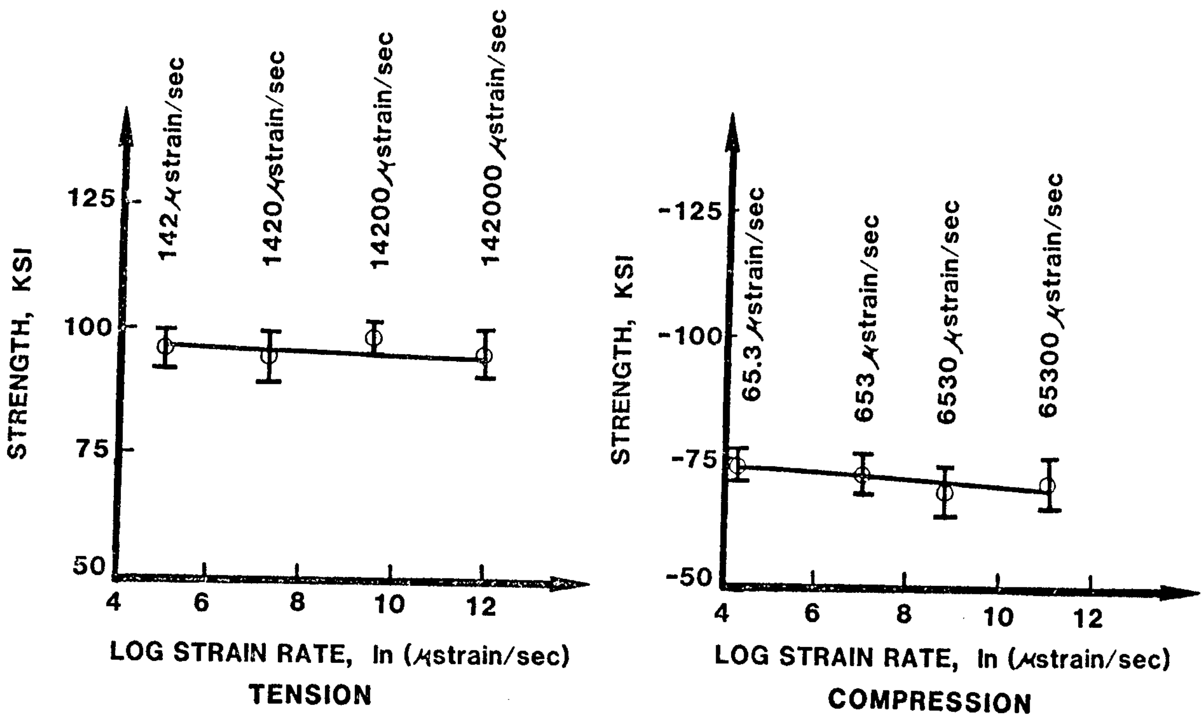


Figure 3 **MEAN STRENGTH ± STANDARD DEVIATION**

LAMINATE 3 (48/48/4) RTD

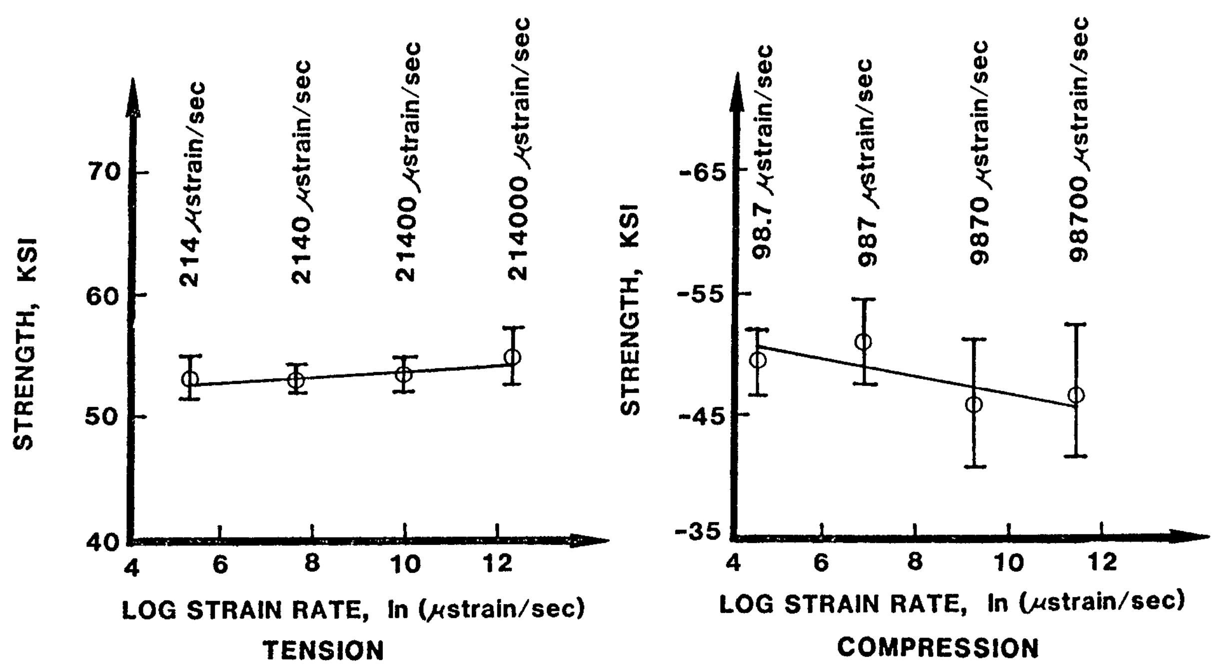


Figure 4 **MEAN STRENGTH ± STANDARD DEVIATION**

LAMINATE 4 (16/80/4) RTD

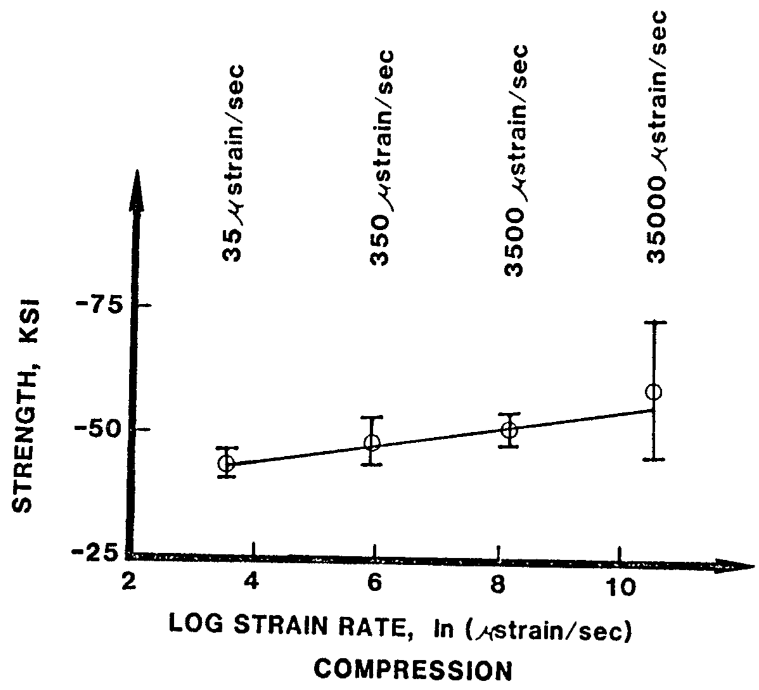


Figure 5 **MEAN STRENGTH ± STANDARD DEVIATION**

LAMINATE 1 (100/0/0) ETW

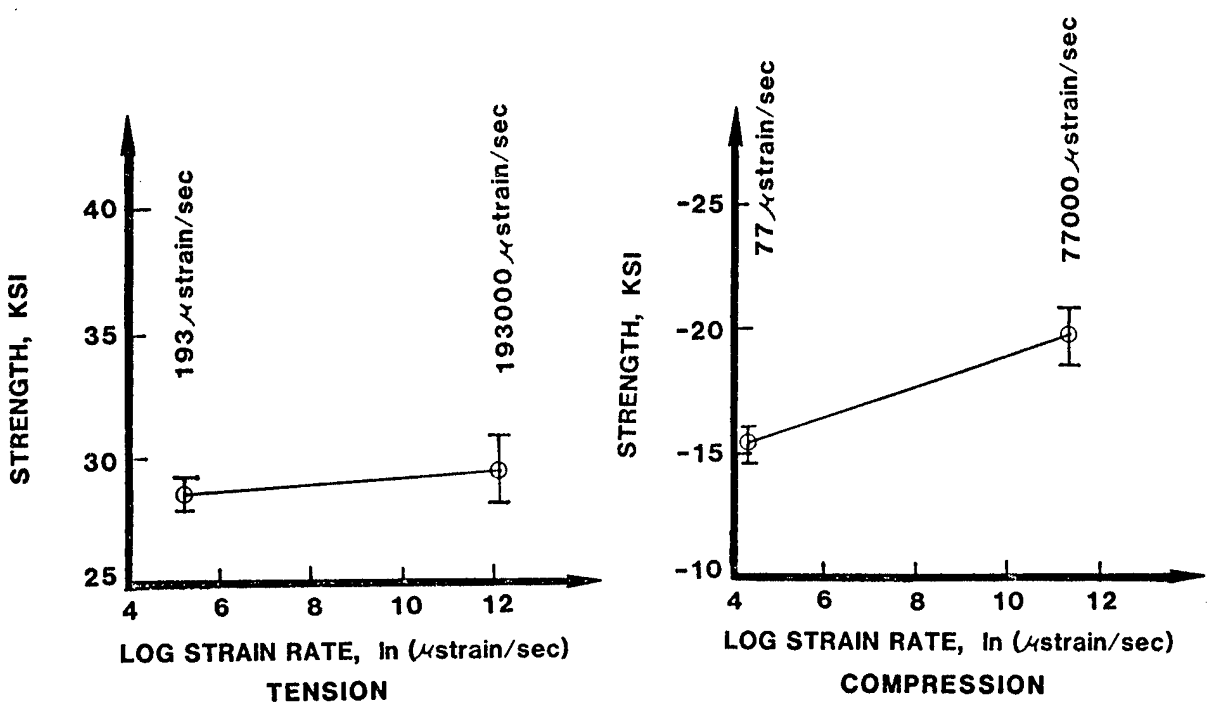


Figure 6 **MEAN STRENGTH ± STANDARD DEVIATION**

LAMINATE 2 (0/100/0) ETW

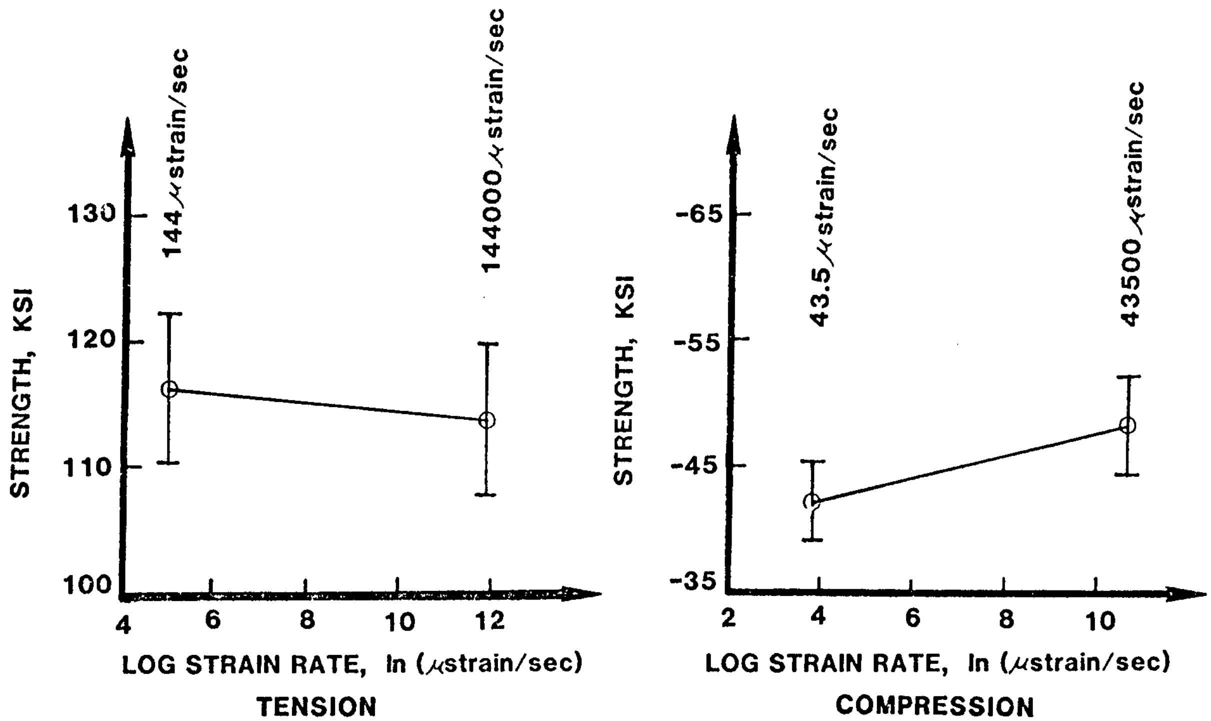


Figure 7 **MEAN STRENGTH±STANDARD DEVIATION**

LAMINATE 3 (48/48/4) ETW

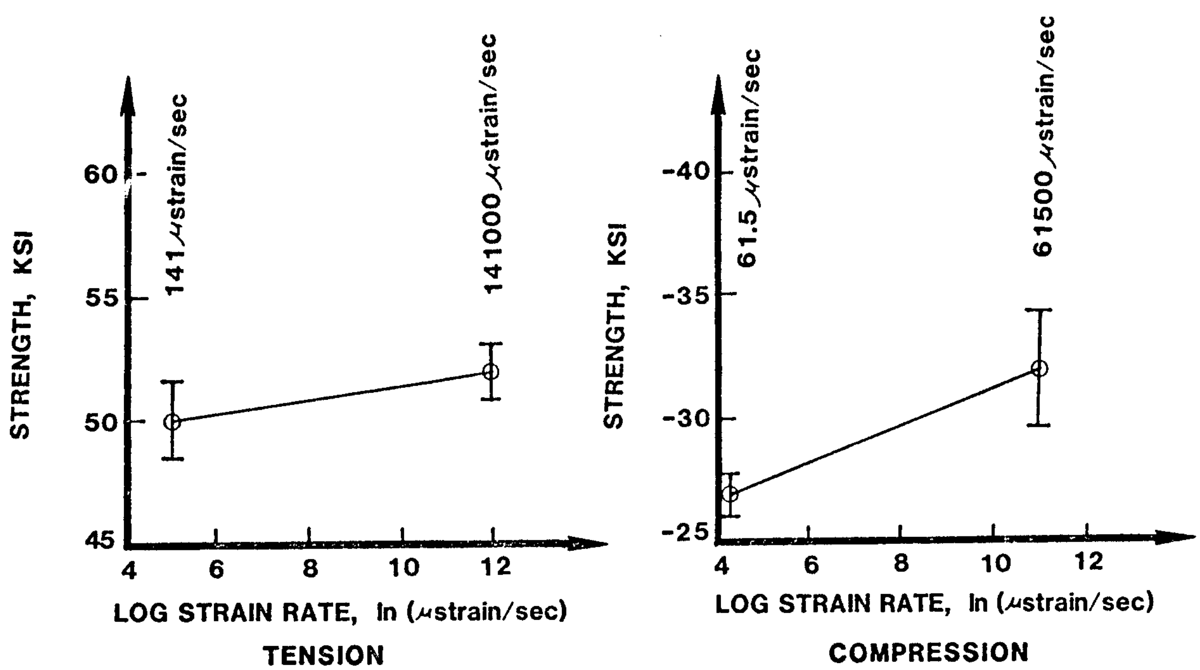


Figure 8 **MEAN STRENGTH±STANDARD DEVIATION**

LAMINATE 4 (16/80/4) ETW

MIXED MODE FRACTURE
OF
UNIDIRECTIONAL COMPOSITES

STEVEN L. DONALDSON
MATERIALS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES

OBJECTIVES

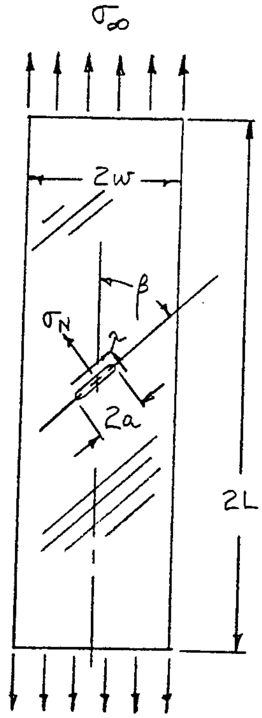
0 EXAMINE THE OFF-AXIS TENSILE TEST AS A METHOD FOR CHARACTERIZING THE TOUGHNESS RESPONSE OF COMPOSITES UNDER COMBINES MODE I AND MODE II LOADING.

0 DEVELOP A COMPARABLE PURE MODE II TEST.

0 UTILIZE THE TESTS TO CHARACTERIZE THE MIXED MODE BEHAVIOR OF A TYPICAL BRITTLE (EPOXY) AND TOUGH (PEEK) MATRIX SYSTEMS.

0 PRELIMINARY SEM EXAMINATION OF FRACTURE SURFACES.

MIXED MODE TEST SPECIMEN
(INCLUDES PURE MODE I, = 90)



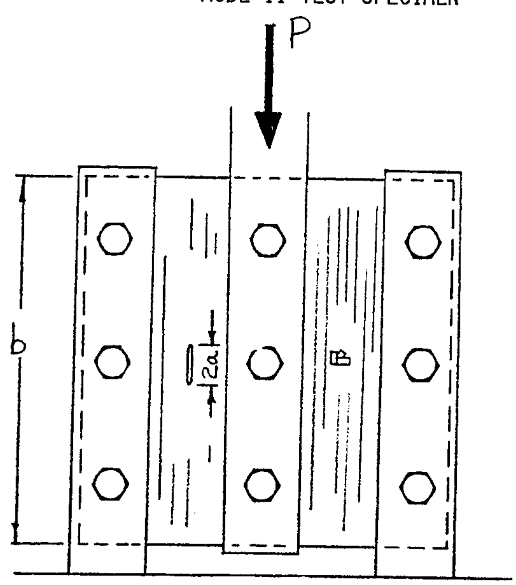
$$K_I = \sigma_N\sqrt{\pi A}$$
$$= \sigma_\infty\sqrt{\pi A}\ \text{SIN}^2\beta$$
$$K_{II} = \tau\sqrt{\pi A}$$
$$= \sigma_\infty\sqrt{\pi A}\ \text{SIN}\beta\ \text{COS}\beta$$

AT CRITICAL (FRACTURE) LOAD:

$$K_{I_C} = \sigma_{\infty_C}\sqrt{\pi A}\ \text{SIN}^2\beta$$

$$K_{II_C} = \sigma_{\infty_C}\sqrt{\pi A}\ \text{SIN}\beta\ \text{COS}\beta$$

MODE II TEST SPECIMEN



$$\tau = \frac{P}{2BH}$$

$$K_{II} = \tau\sqrt{\pi A}$$

AT CRITICAL (FRACTURE) LOAD:

$$K_{II_C} = \tau_C\sqrt{\pi A}$$

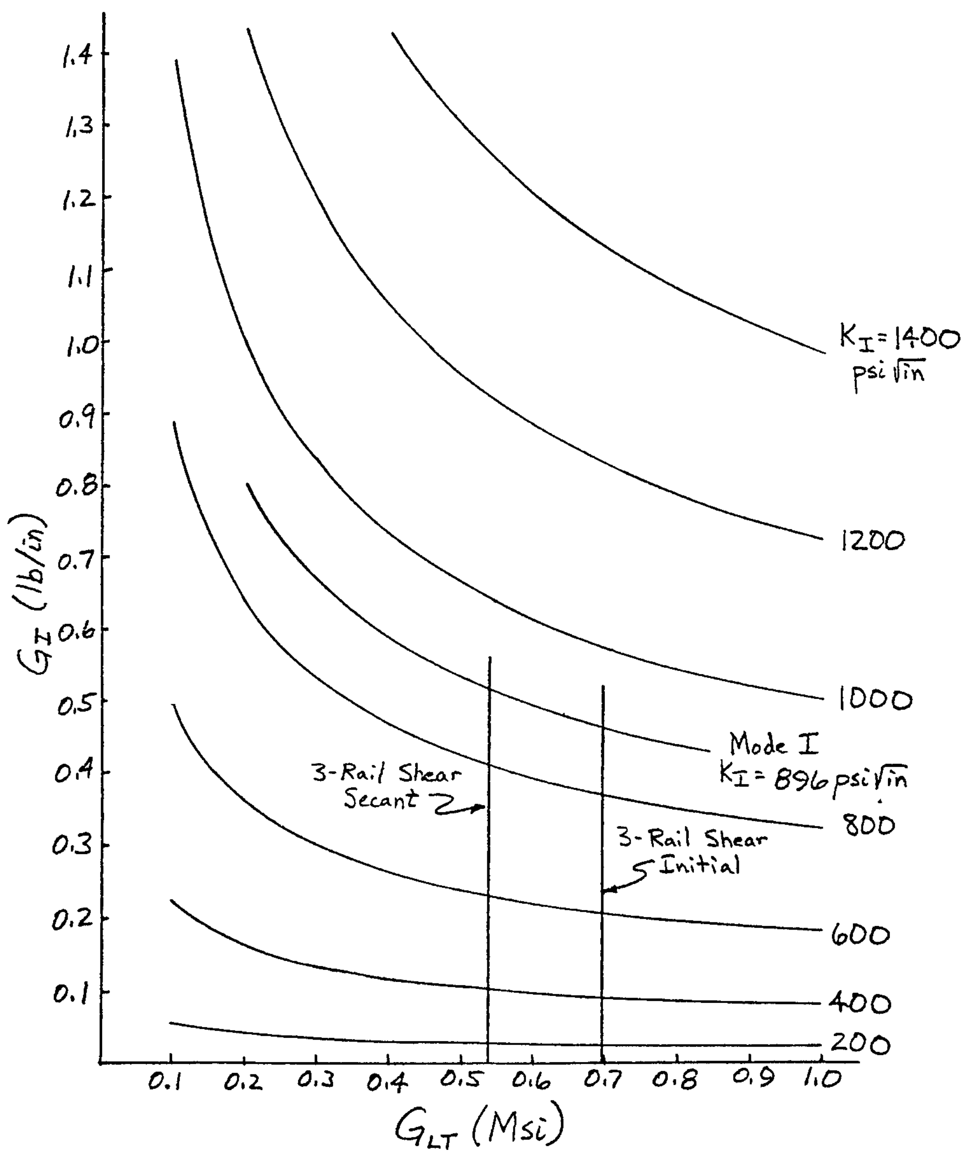
STRESS INTENSITY FACTOR VS. STRAIN ENERGY RELEASE RATE

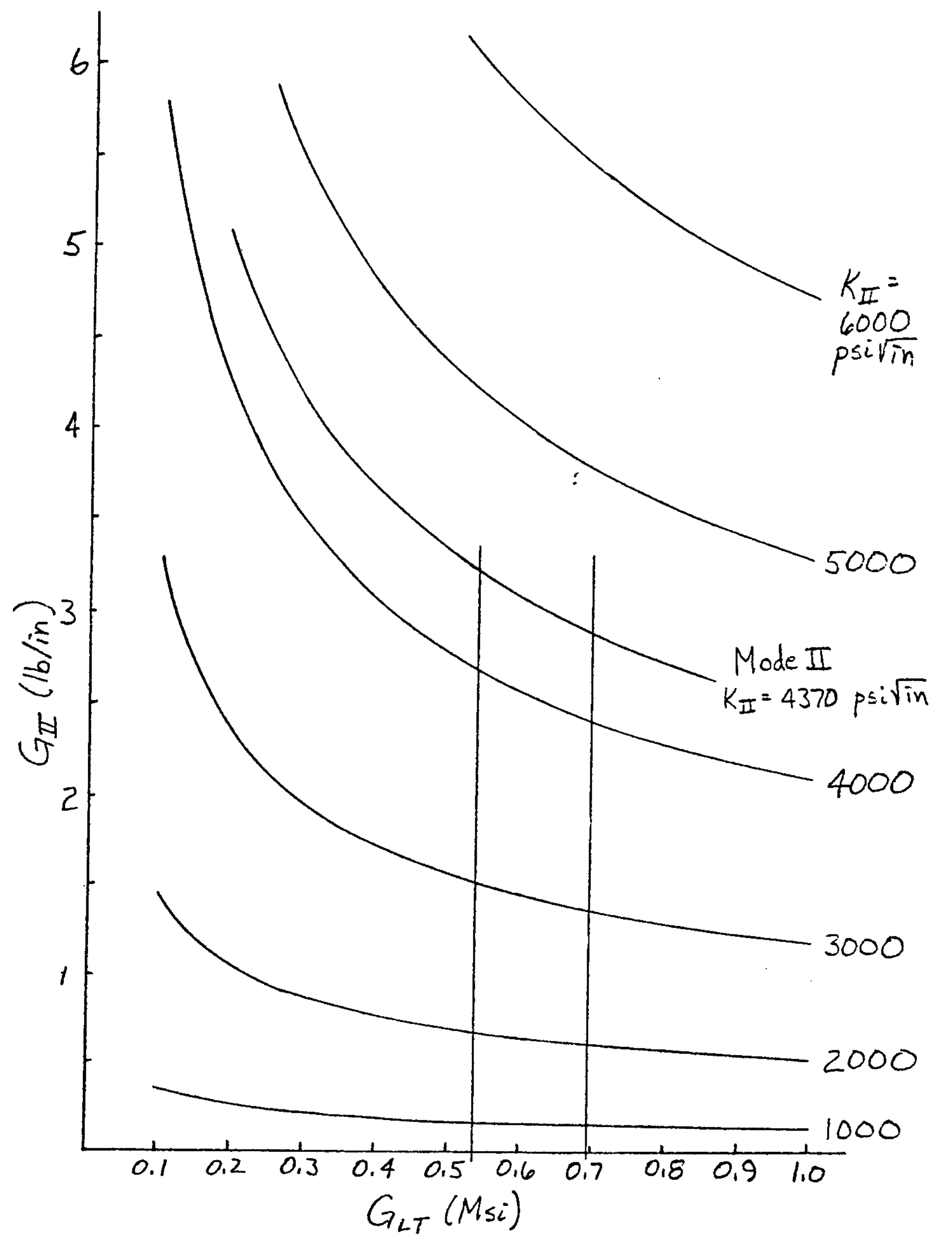
$$G_I = K_I^2 \left[\frac{1}{2E_L E_T}\right]^{\frac{1}{2}} \left[\left(\frac{E_L}{E_T}\right)^{\frac{1}{2}} - \nu_{LT} + \frac{E_L}{2G_{LT}}\right]^{\frac{1}{2}}$$

$$G_{II} = K_{II}^2 \frac{1}{\sqrt{2}\,E_L} \left[\left(\frac{E_L}{E_T}\right)^{\frac{1}{2}} - \nu_{LT} + \frac{E_L}{2G_{LT}}\right]^{\frac{1}{2}}$$

DETERMINE SENSITIVITY OF G_I AND G_{II} TO

K_I, K_{II}, E_L, E_T, ν_{LT}, G_{LT}





POSSIBLE FRACTURE CRITERIA

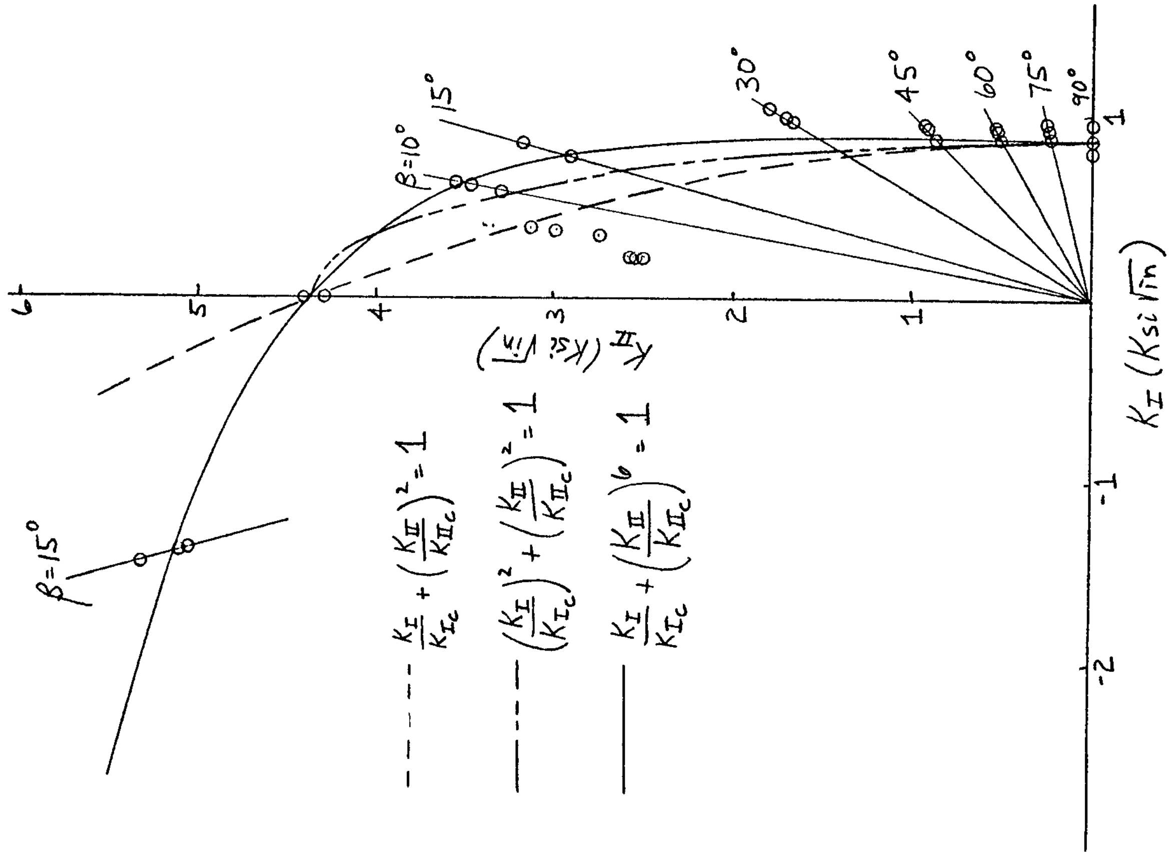
W_N: $\quad \dfrac{K_I}{K_{I_C}} + \left(\dfrac{K_{II}}{K_{II_C}}\right)^2 = 1 \quad ; \quad \left(\dfrac{G_I}{G_{I_C}}\right)^{1/2} + \dfrac{G_{II}}{G_{II_C}} = 1$

$G_I + G_{II}$ = CONST AT FRACTURE = $G_{(I,\ II)_C}$ $\quad ; \quad \left(\dfrac{K_I}{K_{(I,\ II)_C}}\right)^2 + \left(\dfrac{K_{II}}{K_{(I,\ II)_C}}\right)^2 = 1$

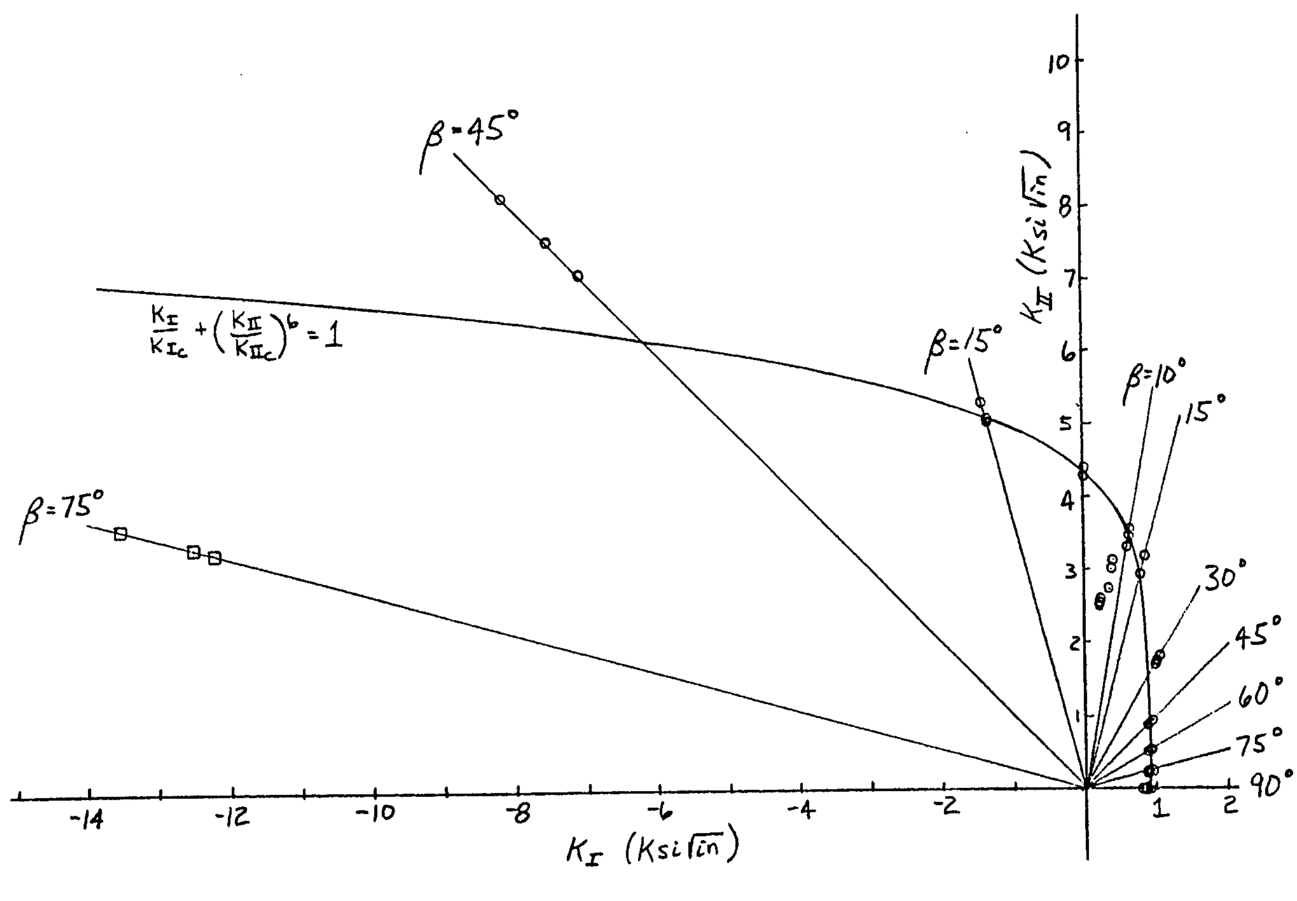
$\dfrac{G_I}{G_{I_C}} + \dfrac{G_{II}}{G_{II_C}} = 1 \quad ; \quad \left(\dfrac{K_I}{K_{I_C}}\right)^2 + \left(\dfrac{K_{II}}{K_{II_C}}\right)^2 = 1$

$\left(\dfrac{G_I}{G_{I_C}}\right)^{M_1} + \left(\dfrac{G_{II}}{G_{II_C}}\right)^{M_2} = 1 \quad ; \quad \left(\dfrac{K_I}{K_{I_C}}\right)^{2M_1} + \left(\dfrac{K_{II}}{K_{II_C}}\right)^{2M_2} = 1$

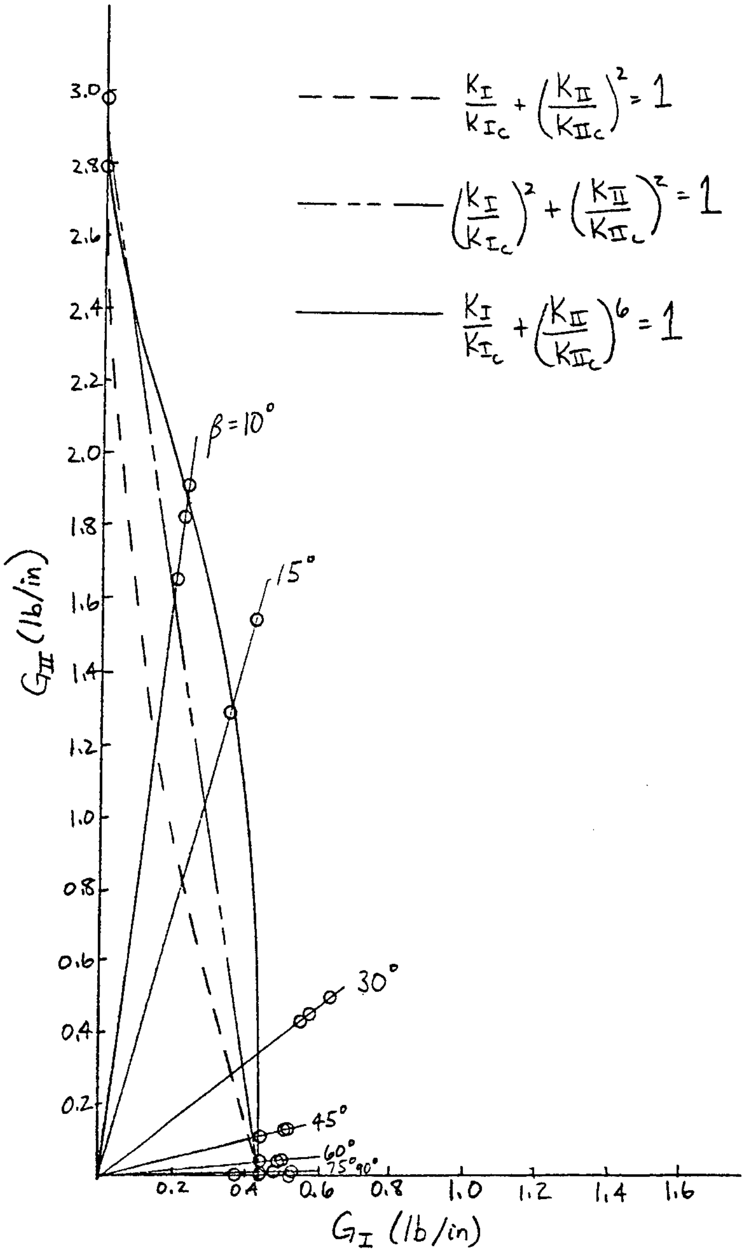
INTERACTION TERMS ?



β=15°
β=10°
15°
30°
45°
60°
75°
90°
$\frac{K_I}{K_{Ic}} + \left(\frac{K_{II}}{K_{IIc}}\right)^2 = 1$
$\left(\frac{K_I}{K_{Ic}}\right)^2 + \left(\frac{K_{II}}{K_{IIc}}\right)^2 = 1$
$\frac{K_I}{K_{Ic}} + \left(\frac{K_{II}}{K_{IIc}}\right)^6 = 1$
K_{II} (Ksi $\sqrt{in}$)
K_I (Ksi $\sqrt{in}$)



$\frac{K_I}{K_{Ic}} + \left(\frac{K_{II}}{K_{IIc}}\right)^6 = 1$
β=45°
β=75°
β=15°
β=10°
15°
30°
45°
60°
75°
90°
K_{II} (Ksi $\sqrt{in}$)
K_I (Ksi $\sqrt{in}$)



CONCLUSIONS:

- 0 OFF-AXIS AND RAIL SHEAR TEST CAN CHARACTERIZE MIXED MODE BEHAVIOR
- 0 K→G CONVERSION SENSITIVE TO G_{LT}, WHICH IS GENERALLY NON-LINEAR
- 0 MODE II DOMINATED BEHAVIOR MOST DIFFICULT TO OBTAIN, YET HIGHEST AREA OF INTERACTION
- 0 $\frac{K_I}{K_{I_C}} + \left(\frac{K_{II}}{K_{II_C}}\right)^6 = 1$ PROVIDES GOOD EMPERICAL FIT TO T300/1034C DATA.

FUTURE WORK/WORK IN PROGRESS:

- 0 NATURAL CRACK BY FATIGUE OR PRE-LOAD
- 0 OFF-AXIS RAIL SHEAR TEST TO OBTAIN MODE II DOMINATED BEHAVIOR
- 0 BOUNDARY EFFECTS:
  - FREE EDGES AND CLAMPED ENDS

SUPPRESSION OF DELAMINATIONS IN COMPOSITES BY THICKNESS DIRECTION REINFORCEMENT

BY

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PRINCIPAL INVESTIGATOR: C.T. SUN

RESEARCH ASSISTANTS: T.M. TAN, L. MIGNERY

OBJECTIVES

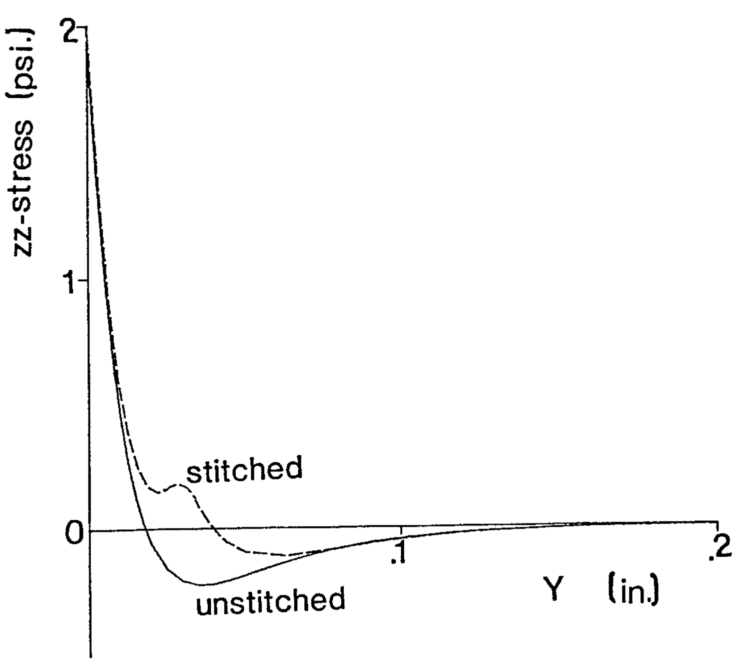
- TO PERFORM ELASTIC STRESS ANALYSIS ON GRAPHITE/EPOXY LAMINATES WITH THICKNESS-DIRECTION REINFORCEMENTS AND INVOKE LEFM TO DETERMINE THE CAPABILITY OF SUCH REINFORCEMENT IN ARRESTING DELAMINATION CRACKS.

- TO CONDUCT STRENGTH AND FATIGUE TESTS ON REINFORCED AND UNREINFORCED LAMINATES.

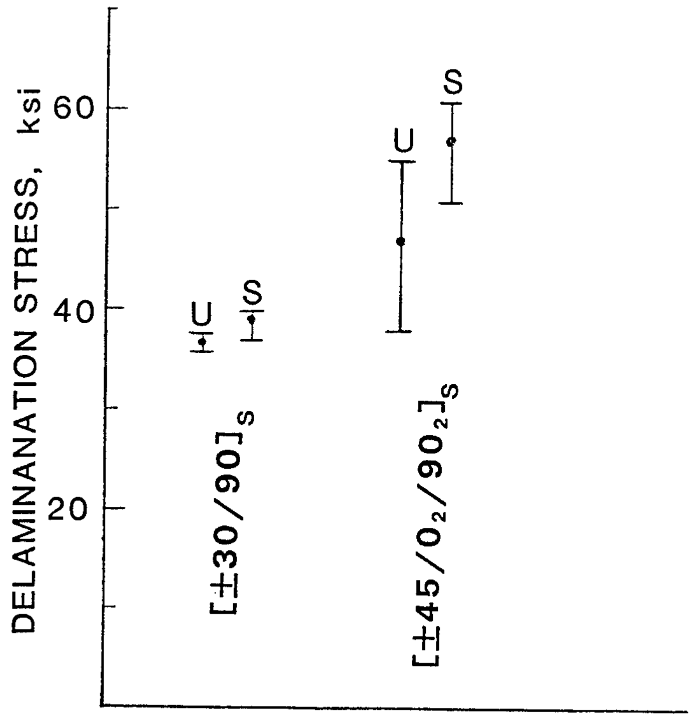
- TO FIND OPTIMAL DESIGN OF THE THICKNESS-DIRECTION REINFORCEMENT.

CONCLUSIONS

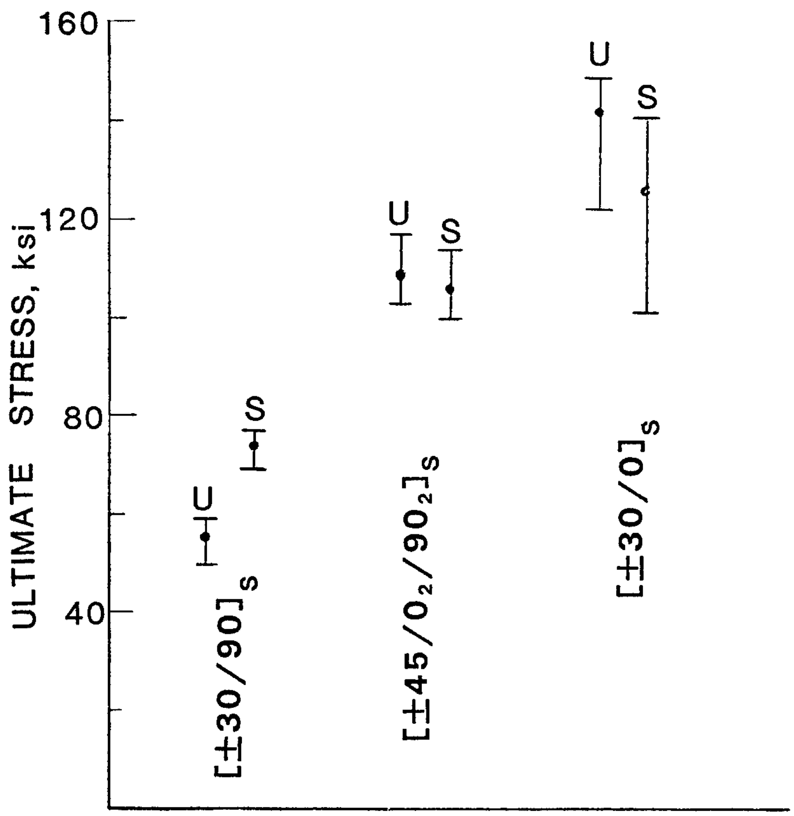
- STITCHING TECHNIQUE IS EFFECTIVE IN SUPPRESSING DELAMINATION
- FOR FIBER-DOMINATED LAMINATES, STITCHING (THUS SUPPRESSION OF DELAMINATION) REDUCES STRENGTH AND FATIGUE LIFE.
- FOR MATRIX-DOMINATED LAMINATES, STITCHING INCREASES BOTH STRENGTH AND FATIGUE LIFE.
- INCLINED STITCHING MAY INCREASE STIFFNESS AND REDUCE INTERLAMINAR SHEAR STRESS.



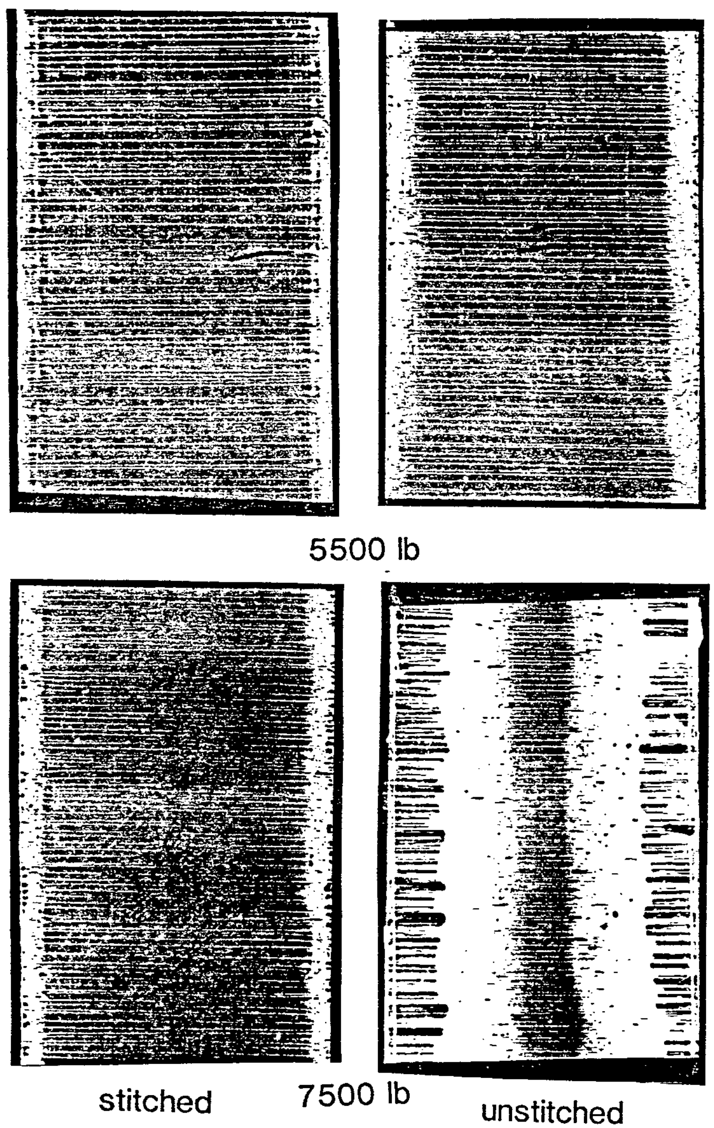
- FREE EDGE NORMAL STRESS IN $[\pm45/0_2/90_2]_S$



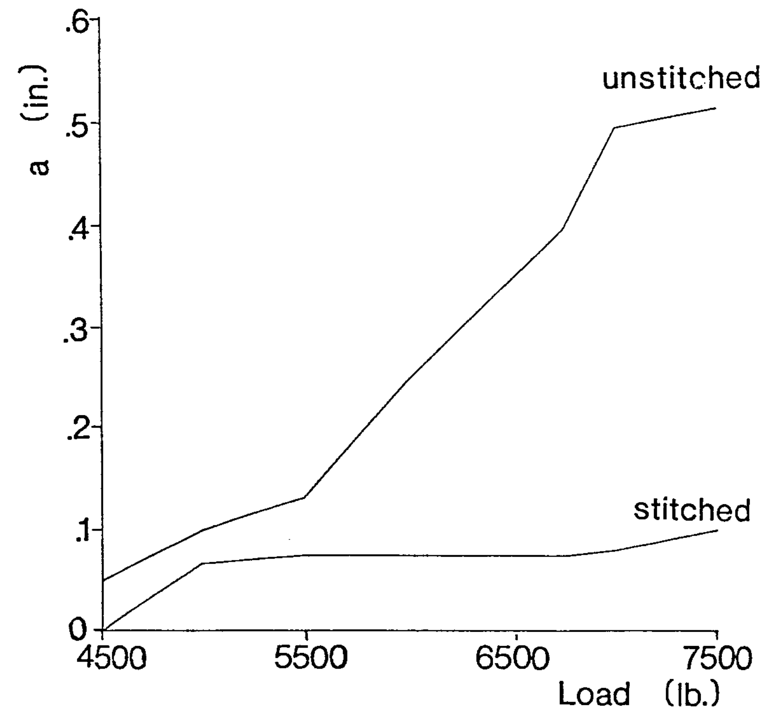
• DELAMINATION STRESS FOR STITCHED AND UNSTITCHED LAMINATES



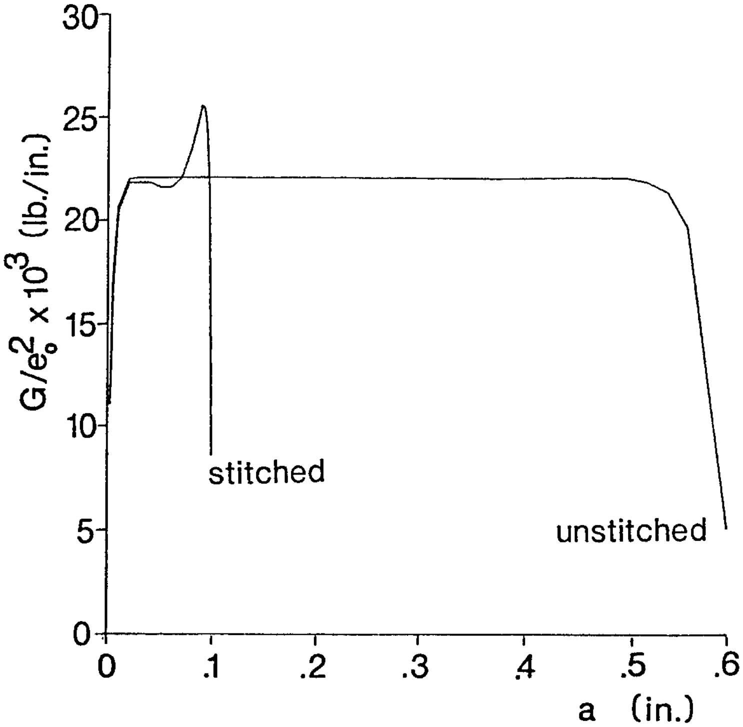
• ULTIMATE STRESS FOR STITCHED AND UNSTITCHED LAMINATES



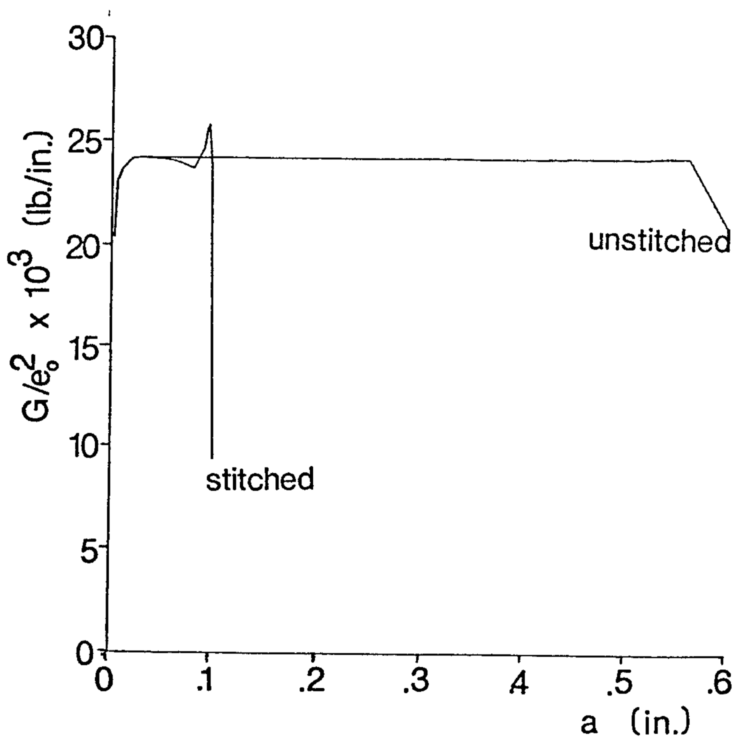
• DELAMINATION IN STITCHED AND UNSTITCHED $[\pm 45/0_2/90_2]_s$ LAMINATES



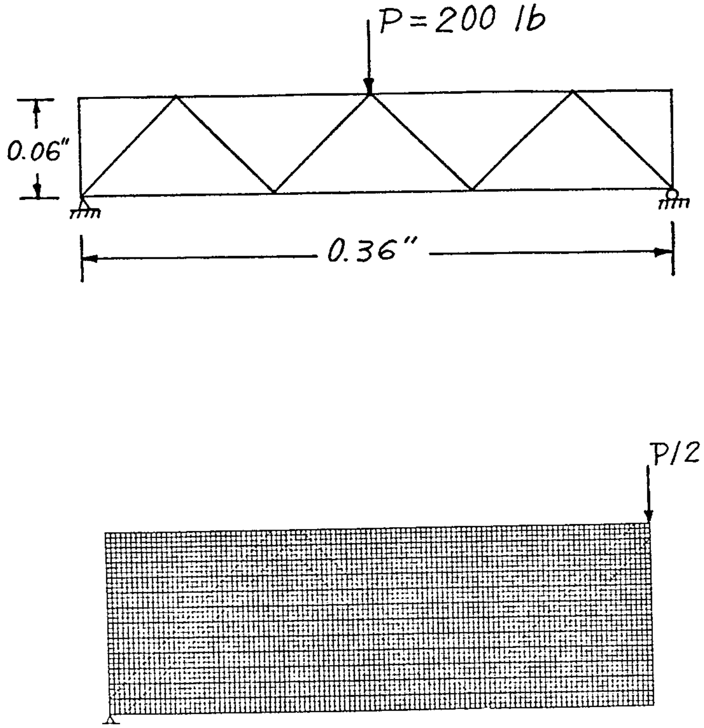
• DELAMINATION CRACK SIZE VS. LOAD FOR $[\pm 45/0_2 90_2]_s$



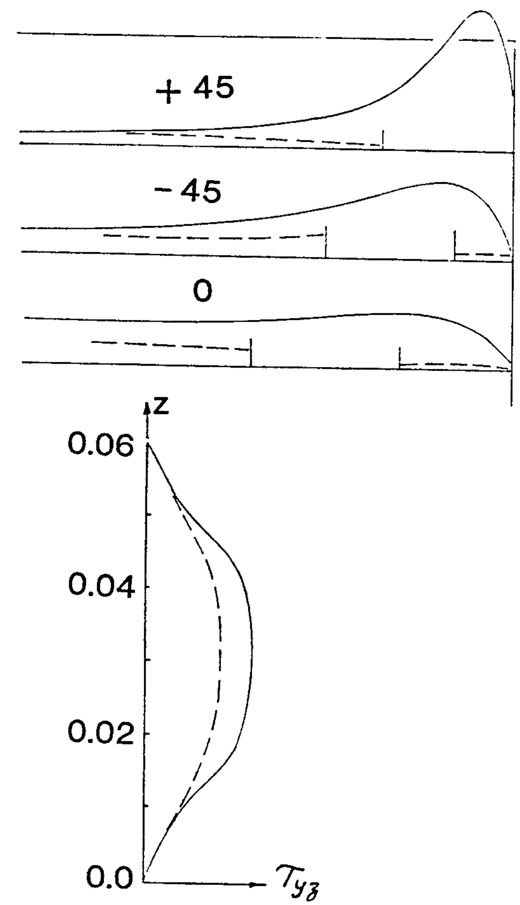
• STRAIN ENERGY RELEASE RATE FOR $[\pm 45/0_2/90_2]_s$



• STRAIN ENERGY RELEASE RATE FOR $[\pm 30/90]_s$



• SHORT BEAM AND FINITE ELEMENT MESH



• INTERLAMINAR SHEAR STRESS IN $[\pm 45/0_2/\mp 45]_s$

ACOUSTIC EMISSION AS AN NDT TOOL FOR COMPOSITES UNDER QUASI-STATIC AND FATIGUE LOADING

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PRESENTED IN THE NINTH ANNUAL MECHANICS OF COMPOSITES REVIEW, DAYTON, OHIO, OCTOBER 24-26, 1983.

GENERAL OBJECTIVES FOR APPLYING ACOUSTIC EMISSION TO COMPOSITE MATERIALS

- DETECT AND LOCATE NON-VISUAL DAMAGE
- DETERMINE DAMAGE INITIATION AND TRACK ITS PROGRESSION
- ANTICIPATE POTENTIAL FRACTURE SITES
- CORRELATE AE RESULTS AND ACTUAL MECHANICAL PROPERTIES, DEFORMATION CHARACTERISTICS AND FRACTURE BEHAVIOR
- IDENTIFY MAJOR FAILURE MECHANISMS, E.G. FIBER FRACTURE, MATRIX CRACKING, DELAMINATION, ETC.
- DETERMINE DAMAGE CRITICALITY
- DISTINGUISH BETWEEN POOR AND GOOD QUALITY MATERIALS, E.G. IDENTIFY FABRICATION INHOMOGENEITIES

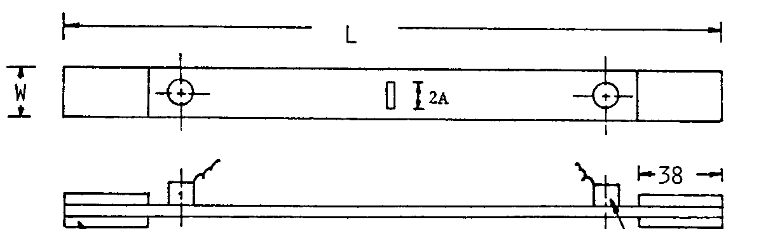
ALL IN REAL-TIME

TEST PROGRAM

TASK I: MONITORING DAMAGE THROUGH AE DURING MONOTONICALLY INCREASING LOAD TO FAILURE.

TASK II: MONITORING DAMAGE THROUGH AE DURING LOADING/UNLOADING CYCLES.

TASK III: MONITORING DAMAGE THROUGH AE DURING FATIGUE LOADING.



W = 25 MM, 51 MM

L = 160 MM, 330 MM

2A/W = 0.0 ⟶ 0.5

MATERIALS: GRAPHITE/EPOXY, GRAPHITE/POLYIMIDE, BORON/ALUMINUM

THE PRESENTATION INCLUDES REPRESENTATIVE RESULTS FROM THE FOLLOWING MATERIAL SYSTEMS

I. GRAPHITE/EPOXY LAMINATES:

UNIDIRECTIONAL OFF-AXIS: 0 = 0°, 2.5°, 5°, 7.5°, 10°, 15°, 20°, ...

CENTER NOTCHED $[0°]_8$: 2 A/W = 0.0, 0.1, 0.3, 0.5

$[\pm 45]_{2S}$: AS FABRICATED

ARTIFICIALLY INDUCED DELAMINATION (TEFLON SPRAY AND FOIL)

$[0/90]_{2S}$, $[(90)_2/(0)_2]_S$, $[\pm 45/90]_{2S}$

CENTER NOTCHED $[0_2/+45/0_2/-45/0/90]_S$: 2A/W = 0.0, 0.1, 0.3,0.5

DOUBLE EDGE NOTCHED FILAMENT-WOUND $[\pm 24/90/0/\pm 45/0/90/\pm 24]_T$:

2A/W = 0.0, 0.05, 0.1, 0.15, 0.20, 0.25

II. GRAPHITE/POLYIMIDE LAMINATES:

CENTER NOTCHED $[0/+45/90/-45]_{2S}$ VS. $[90/+45/0/-45]_{2S}$

$[0/90]_{2S}$, $[\pm 45]_{2S}$, $[0°]_8$: 2A/W = 0.0, 0.02, 0.05, 0.1, 0.2, 0.3, 0.4, 0.5

III. BORON/ALUMINUM LAMINATES:

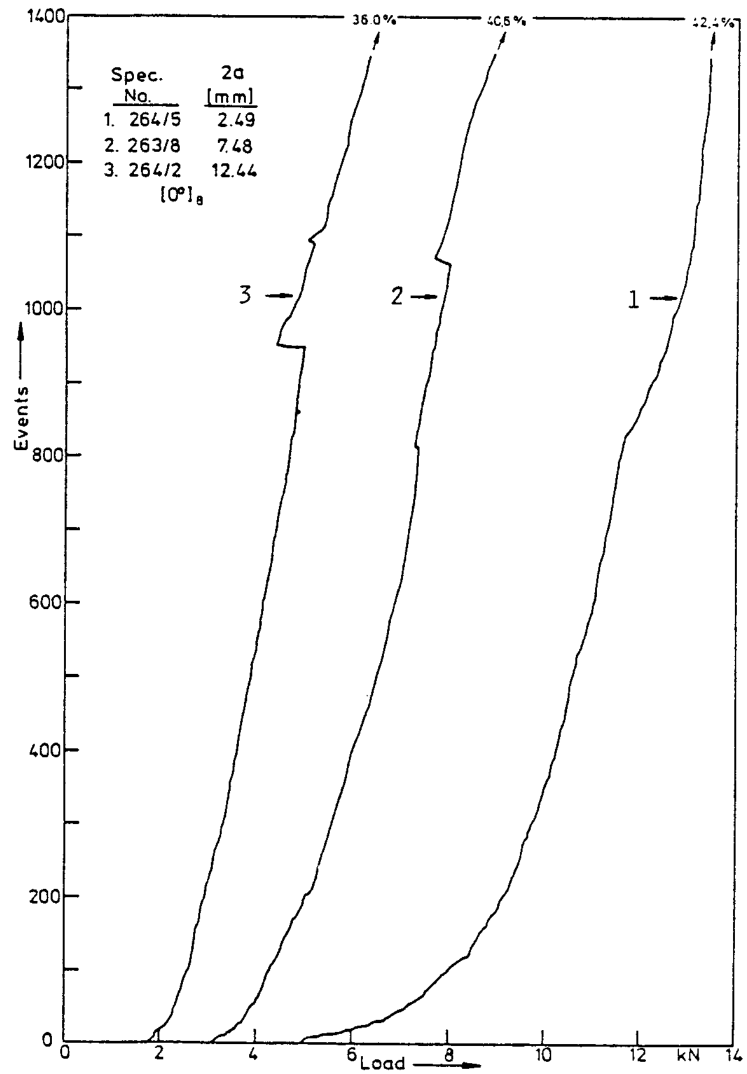
CENTER NOTCHED: $[0]_8$, $[90]_8$, $[0/90]_{2S}$, $[\pm 45]_{2S}$, $[0_2/\pm 45]_S$, $[0/\pm 45/90]_S$; 2A/W = 0.0, 0.05, 0.1, 0.2, 0.3, 0.4, 0.5, AND CONSTITUENTS

IV. TEST PARAMETERS·

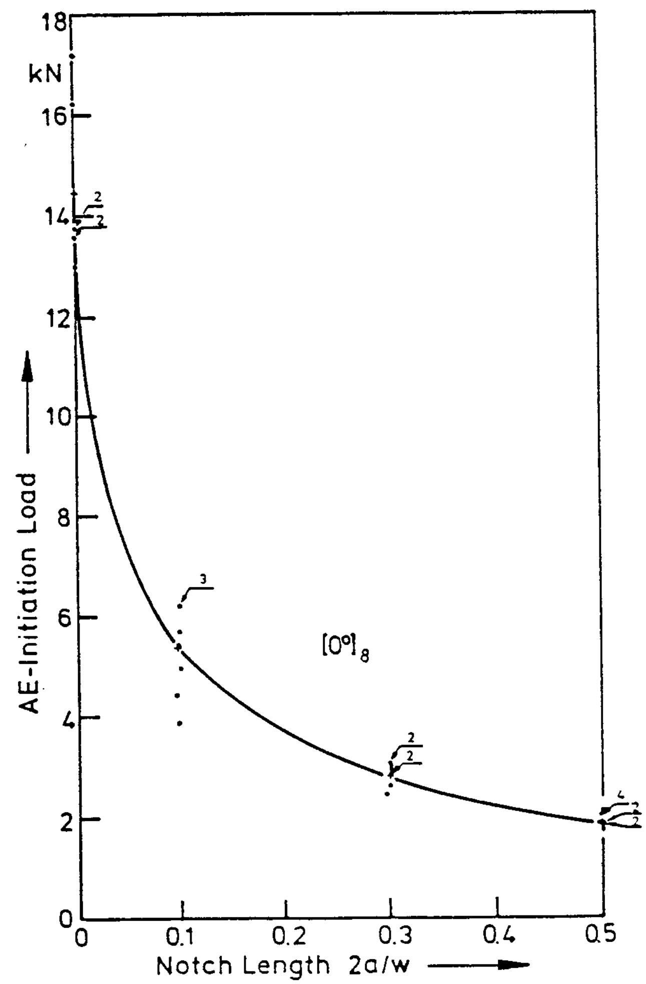
EFFECT(S) OF TYPES OF AE TRANSDUCERS

EFFECT(S) OF AE INSTRUMENTATION PARAMETERS

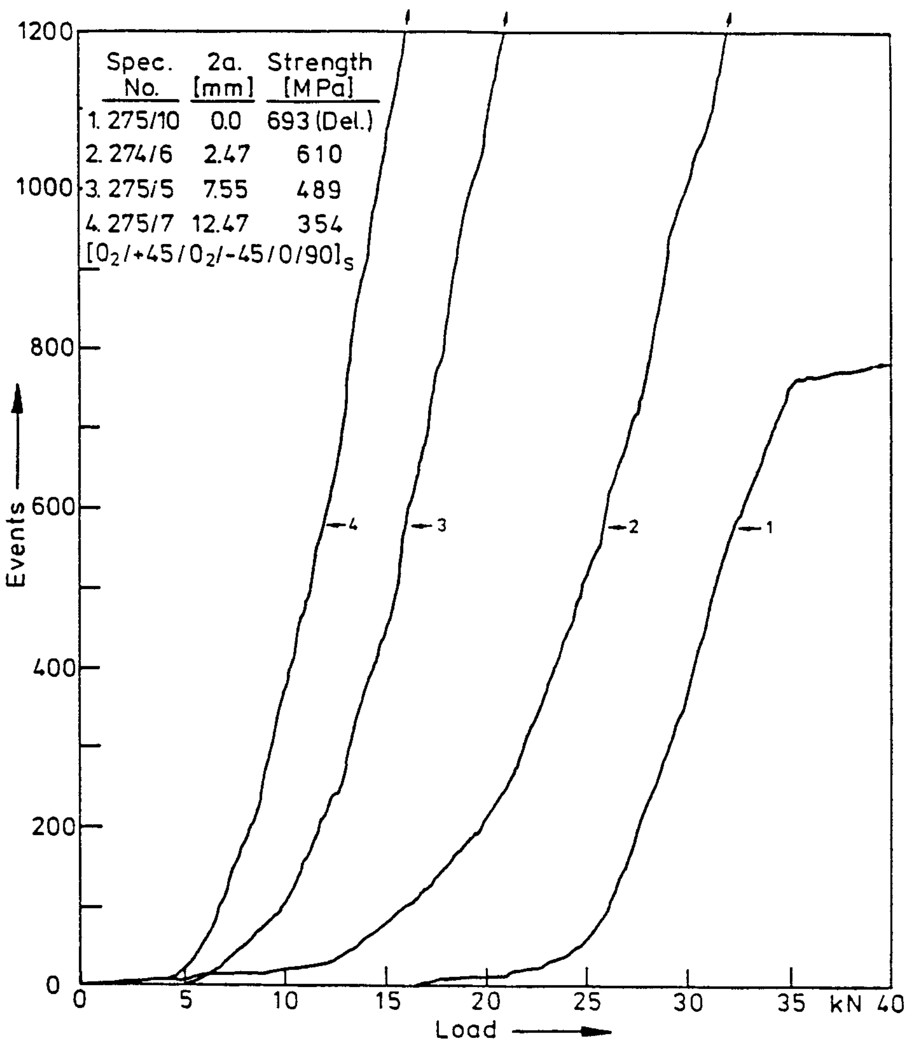
EFFECT(S) OF TESTING PROCEDURES



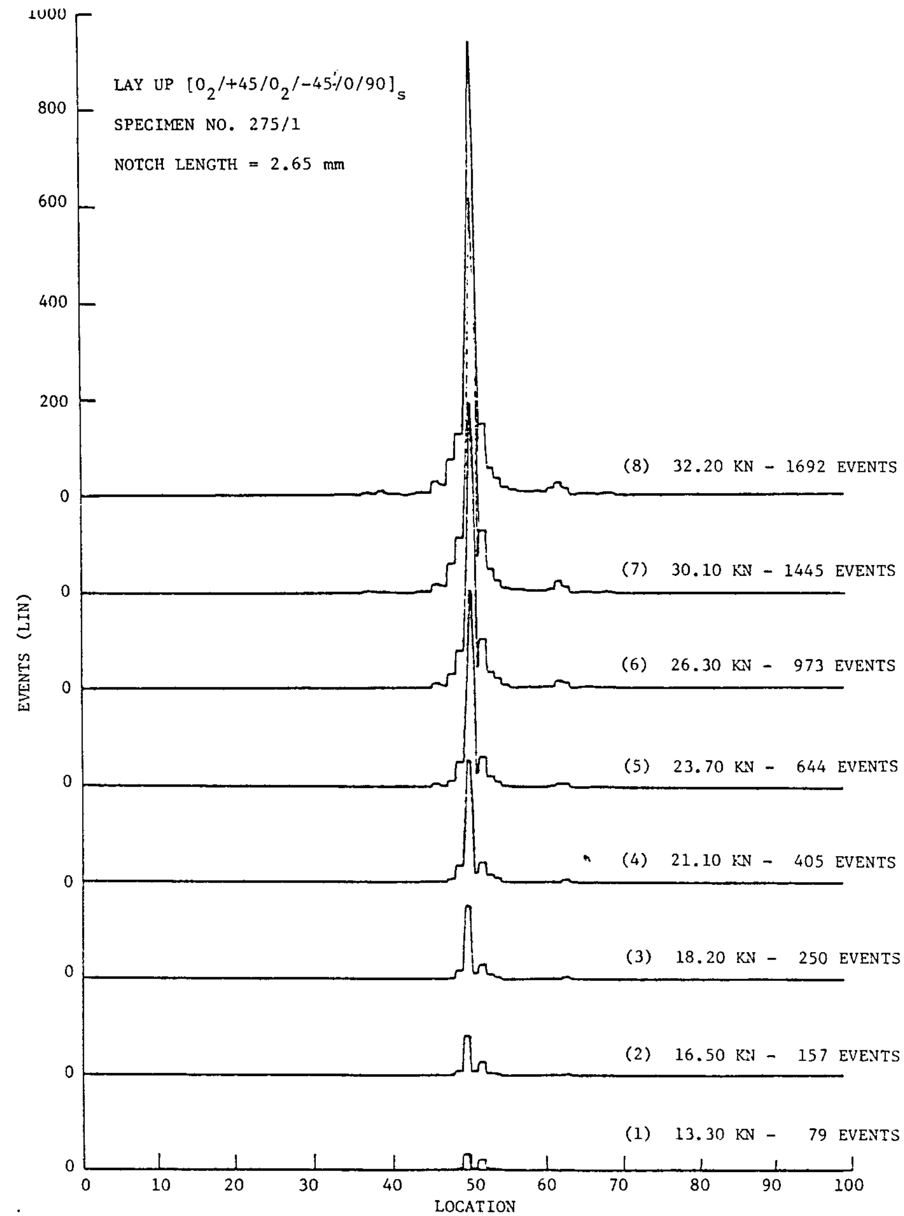
ACCUMULATIVE EVENTS AS A FUNCTION OF LOAD FOR CENTER-NOTCHED GR/EP $[0]_8$, SHOWING THAT EMISSION INITIATION LOAD DEPENDS ON DAMAGE SIZE.



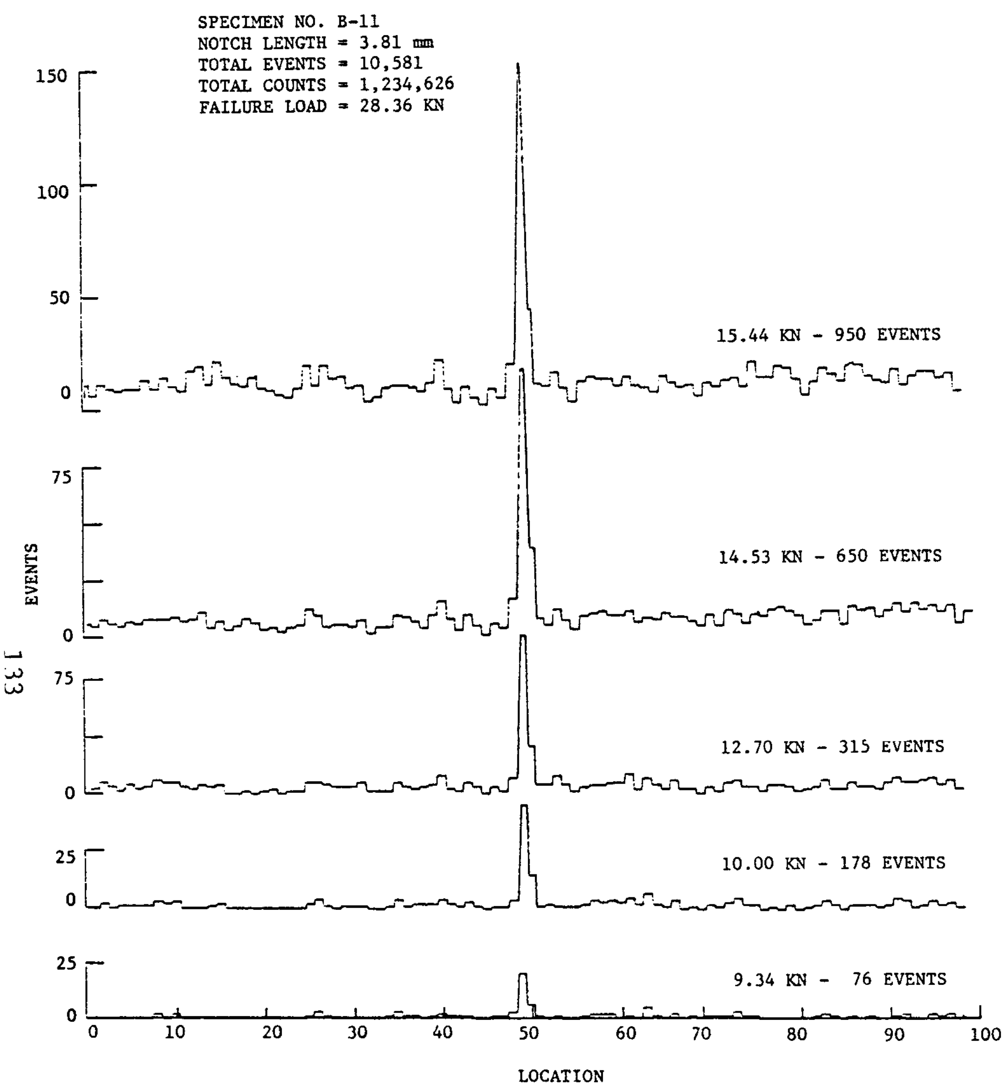
EMISSION INITIATION LOAD AS A FUNCTION OF NOTCH LENGTH FOR GR/EP $[0]_8$.



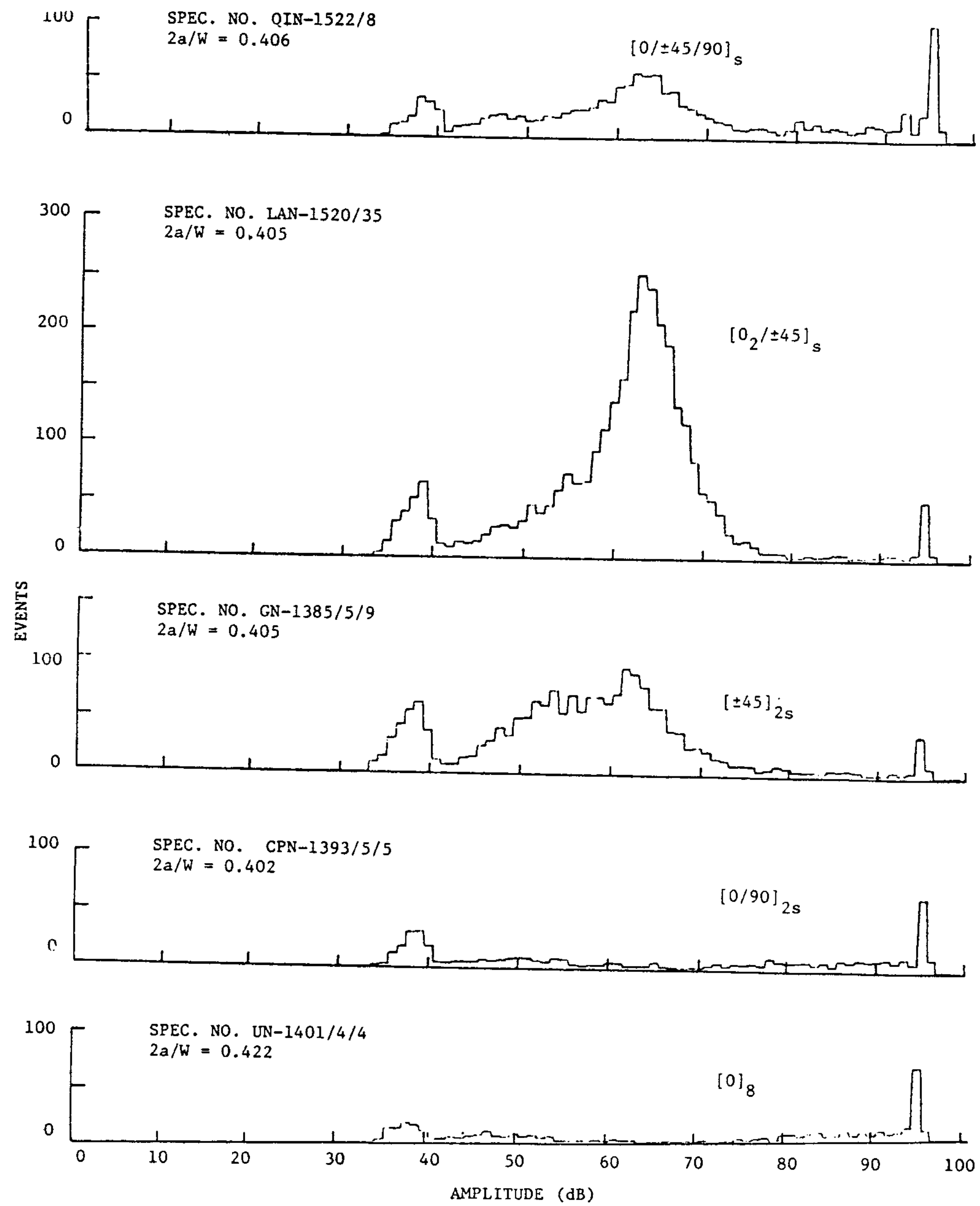
ACCUMULATIVE EVENTS AS A FUNCTION OF LOAD FOR CENTER-NOTCHED GR/EP $[0_2/+45/0_2/-45/0/90]_s$ LAMINATE, SHOWING THE EFFECT OF NOTCH LENGTH ON AE RESULTS.



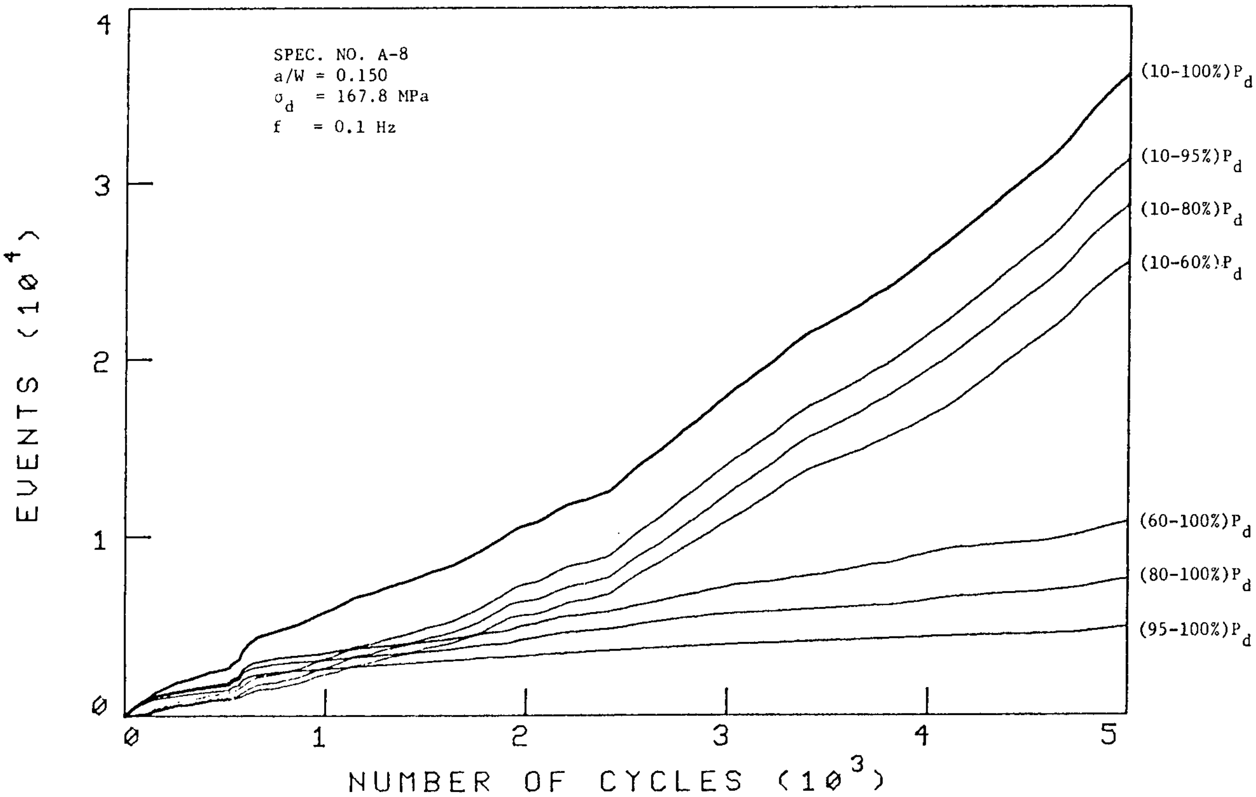
LOCATION DISTRIBUTION HISTOGRAMS OF EVENTS FOR CENTER-NOTCHED GR/EP $[0_2/+45/0_2/-45/0/90]_s$ LAMINATE AT VARIOUS LOAD LEVELS. NOTCH (LOCATED AT LOCATION "50") CAN BE EASILY DETECTED AND LOCATED.



A) LOCATION DISTRIBUTION HISTOGRAMS OF EVENTS FOR DOUBLE-EDGE NOTCHED FILAMENT-WOUND GR/EP $[\pm24/90/0/\pm45/0/90/\pm24]_T$ LAMINATE. NOTCH (LOCATED AT LOCATION "50") CAN BE EASILY DETECTED AND LOCATED, HOWEVER SIGNIFICANT EMISSION IS DETECTED THROUGHOUT THE SPECIMEN LENGTH PRIMARILY DUE TO THE LARGE VOID CONTENT.



AMPLITUDE DISTRIBTUION HISTOGRAMS FOR VARIOUS CENTER-NOTCHED BORON/ ALUMINUM LAMINATES. HIGH AMPLITUDE EVENTS CORRESPOND TO FIBER BREAKAGE.



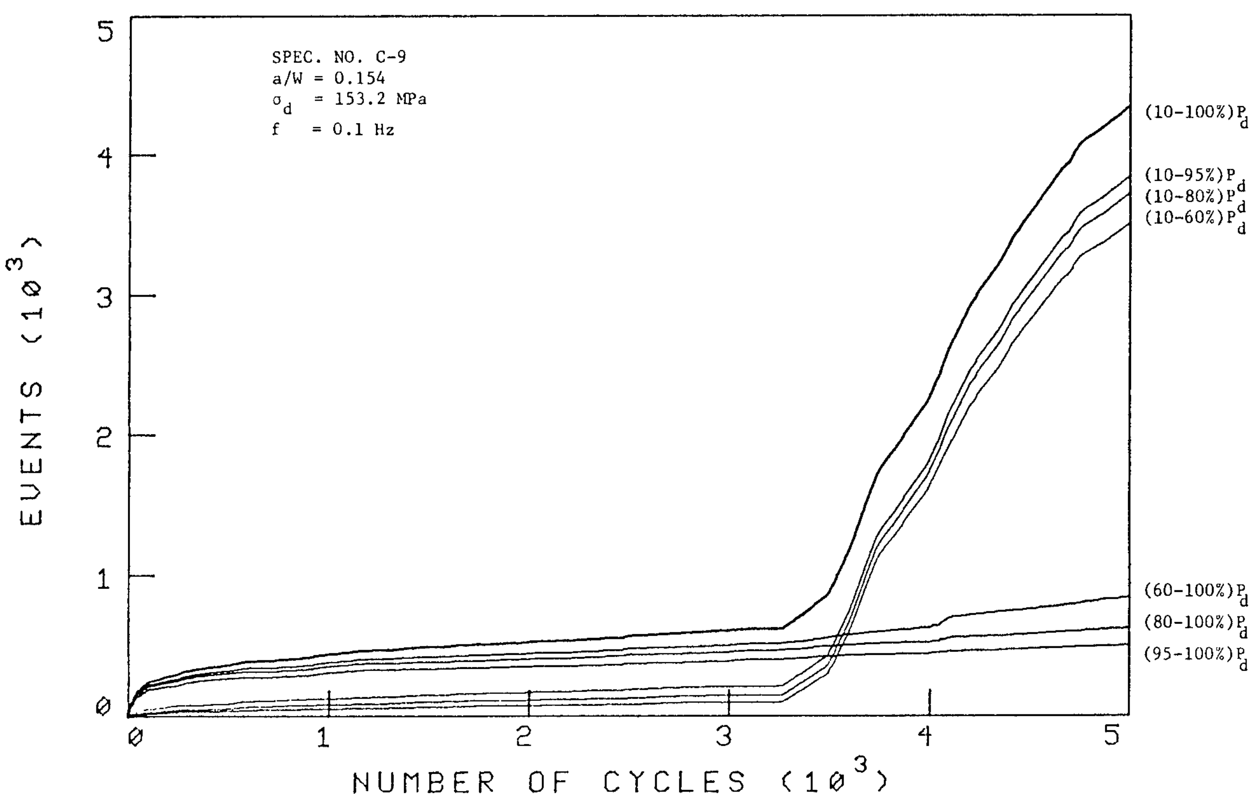
ACCUMULATIVE EVENTS AS A FUNCTION OF NUMBER OF CYCLES FOR DOUBLE-EDGE NOTCHED FILAMENT-WOUND GR/EP $[\pm24/90/0/\pm45/0/90/\pm24]_T$ LAMINATE (NOTCH LENGTH-TO-WIDTH RATIO = 0.15), DISTINGUISHING EMISSION GENERATED IN DIFFERENT LOAD RANGES.

CONCLUSIONS - TASK I

- FROM LOCATION DISTRIBUTION HISTOGRAMS OF EVENTS, ARTIFICIALLY INDUCED DAMAGE CAN BE EASILY DETECTED AND LOCATED AND DAMAGE PROGRESSION CAN BE TRACKED.
- LOCATION DISTRIBUTION HISTOGRAMS OF EVENTS CAN IDENTIFY MATERIAL QUALITY.
- QUALITY OF MATERIAL CAN BE DETERMINED FROM EMISSION INITIATION LOAD.
- EVENT AMPLITUDE LEVELS CAN DISTINGUISH BETWEEN FIBER FAILURE AND MATRIX DOMINATED FAILURES:
  - IN RESIN MATRIX COMPOSITES, DISTINCTION AMONG THE VARIOUS MATRIX-DOMINATED FAILURE MODES, E.G. MATRIX CRACKING, DELAMINATION AND MICRO-FAILURE MECHANISMS, IS QUESTIONABLE.
  - IN METAL MATRIX COMPOSITES, IT SEEMS THAT SUCH A DISTINCTION CAN BE MADE, E.G. BETWEEN INTERFACIAL FAILURE AND MATRIX PLASTIC DEFORMATION.
- ACOUSTIC EMISSION RESULTS (E.G. NUMBER OF EVENTS, COUNTS, ETC.) HAVE PRIMARILY QUALITATIVE RATHER THAN QUANTITATIVE SIGNIFICANCE.
- THESE QUALITATIVE VALUES VARY SIGNIFICANTLY WITH MATERIAL CHARACTERISTICS, E.G, CONSTITUENTS, LAMINATE CONFIGURATION AND GEOMETRY, QUALITY OF FABRICATION, MECHANICAL PROPERTIES, FRACTURE BEHAVIOR, NOTCH SENSITIVITY, FAILURE MODES, ETC., AND THEY STRONGLY DEPEND ON EXTRINSIC VARIABLES (LOADING FUNCTIONS, ETC.) AND ON AE INSTRUMENTATION PARAMETERS (TRANSDUCERS, FILTERS, THRESHOLD LEVELS, ETC.).

CONCLUSIONS TASK-III

- POST-FATIGUE EMISSION INITIATION LOAD IS SLIGHTLY HIGHER THAN THE MAXIMUM FATIGUE STRESS.
- POST-FATIGUE EMISSION INITIATION LOAD IS SIGNIFICANTLY HIGHER THAN PRE-FATIGUE INITIATION LOAD, AND DEPENDS STRONGLY ON FATIGUE STRESS LEVEL. CONSEQUENTLY, LOAD HISTORY IS A SIGNIFICANT PARAMETER IN EVALUATING MATERIAL QUALITY AND/OR DAMAGE PROGRESSION THROUGH ACOUSTIC EMISSION.
- THERE IS A LARGE SCATTER IN THE EVENTS-VERSUS-NUMBER-OF-CYCLES PLOTS AMONG THE DIFFERENT SPECIMENS, INDICATING A LARGE VARIATION IN DAMAGE PROGRESSION AND MATERIAL QUALITY.
- A SIGNIFICANT AMOUNT OF EMISSION IS GENERATED BY FRICTION AMONG FRACTURE SURFACES WHICH HAVE BEEN CREATED DURING THE FATIGUE LOADING, E.G. MATRIX CRACKING AND DELAMINATION. IT IS ESSENTIAL TO DIFFERENTIATE BETWEEN THIS TYPE OF EMISSION AND EMISSION GENERATED BY ACTUAL DAMAGE PROGRESSION.
- DURING THE INITIAL FATIGUE LOADING MOST OF THE EMISSION SEEMS TO BE GENERATED BY DAMAGE PROGRESSION. WITH INCREASING NUMBER OF CYCLES, MOST OF THE EMISSION IS PRIMARILY DUE TO FRICTION.



ACCUMULATIVE EVENTS AS A FUNCTION OF NUMBER OF CYCLES FOR DOUBLE-EDGE NOTCHED FILAMENT-WOUND GR/EP [±24/90/0/±45/0/90/±24]$_T$ LAMINATE (NOTCH LENGTH-TO-WIDTH RATIO = 0.15), DISTINGUISHING EMISSION GENERATED IN DIFFERENT LOAD RANGES.

- FRICTION-GENERATED EMISSION SHOULD NOT BE ELIMINATED FROM THE RECORDED INFORMATION BECAUSE IT CAN SERVE AS AN IMPORTANT INDICATOR OF DAMAGE PROGRESSION. MATRIX CRACKING PROGRESSES SO RAPIDLY THAT THE AE INSTRUMENTATION MAY FAIL TO REGISTER THE ASSOCIATED EVENTS; HOWEVER, THE SUDDEN RESULTING INCREASES IN FRICTION-GENERATED EMISSION INDICATE THAT DAMAGE PROGRESSION HAS OCCURRED.

- A DIRECT CORRELATION HAS BEEN CLEARLY ESTABLISHED BETWEEN THE AE RESULTS WHICH INDICATE FRICTION-GENERATED EMISSION AND THE VISUAL OBSERVATIONS OF DAMAGE PROGRESSION MADE THROUGH THE CCTV.

- A DIRECT CORRELATION COULD BE ESTABLISHED BETWEEN LOCATION DISTRIBUTION HISTOGRAMS OF EVENTS AND THE EVENTS-VERSUS-NUMBER-OF-CYCLES PLOT.

- THE CYCLE NUMBER AT WHICH A SUDDEN DAMAGE GROWTH HAS OCCURRED CAN BE EASILY AND PRECISELY DETERMINED AND CAN ALSO INDICATE THE LOCATION AT WHICH THE DAMAGE HAS OCCURRED. CONSEQUENTLY, THE MONITORING OF ACOUSTIC EMISSION DURING FATIGUE LOADING CAN SAVE TIME AND MONEY WHEN STUDYING FATIGUE DAMAGE INITIATION AND PROGRESSION IN COMPOSITE SYSTEMS.

Mechanical Characterization of "Magnaweave" Braided Composites

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NAVAL AIR DEVELOPMENT CENTER
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OBJECTIVE

QUANTIFY POSSIBLE ADVANTAGES OF FULLY-INTEGRATED BRAIDED COMPOSITES TO OVERCOME LIMITATIONS OF CURRENT LAMINATED COMPOSITES

- IMPACT
- SHORT TRANSVERSE STRENGTH
- DELAMINATION

CHARACTERIZE MECHANICAL PROPERTIES OF BRAIDED COMPOSITES AND DETERMINE THEIR SUITABILITY FOR APPLICATION ON AIRCRAFT STRUCTURES

"MAGNAWEAVE"

GENERAL BRAIDING PROCESS:

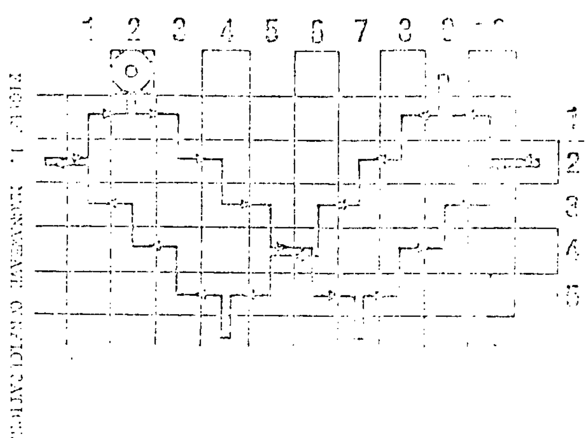
INTERLACING AND ORIENTATION OF YARNS IS ACHIEVED BY ORTHOGONAL SHEDDING MOTION FOLLOWED BY COMBING MOTION.

ADVANTAGES:

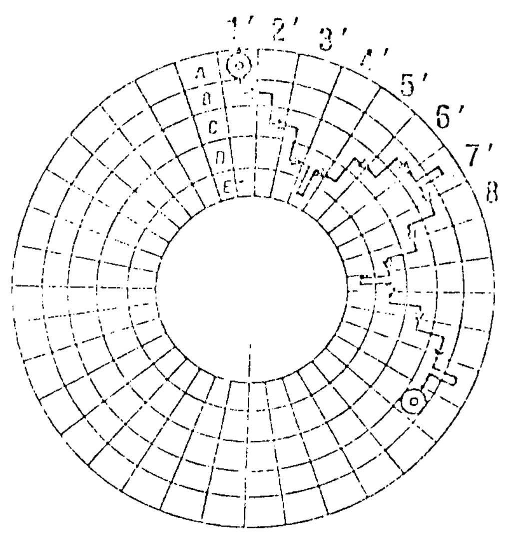
FREEDOM IN MATERIAL DISTRIBUTION AND ORIENTATION.

INTERGRATED STRUCTURAL GEOMETRY

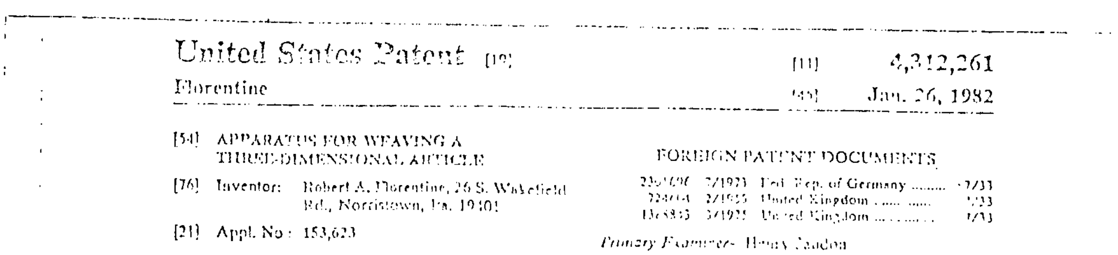
CAN ASSUME COMPLEX SHAPES



MAGNA WEAVE



MAGNA SWIRL



United States Patent [19]
Florentine

[11] 4,312,261
[45] Jan. 26, 1982

[54] APPARATUS FOR WEAVING A THREE-DIMENSIONAL ARTICLE

[76] Inventor: Robert A. Florentine, 26 S. Wakefield Rd., Norristown, Pa. 19401

[21] Appl. No.: 153,623

FOREIGN PATENT DOCUMENTS

[illegible] Fed. Rep. of Germany
[illegible] United Kingdom
[illegible] United Kingdom

Primary Examiner—[illegible]

TEST SPECIMENS

THICKNESS = .125 in.

BASELINE

AS/3501

(+45/−45/0/0/+45/−45/0/0/+45/−45/0/90)s

MAGNAWEAVE

C12000/3501

(1X1)BRAID

C12000/3501

(1X1)ADDED LONGITUDINAL

"MAGNAWEAVE" PROBLEMS

RESIN IMPREGNATION

FIBER VOLUME CONTROL

THICKNESS CONTROL

SURFACE TEXTURE

MICRO—CRACKS

ANALYSIS METHODS

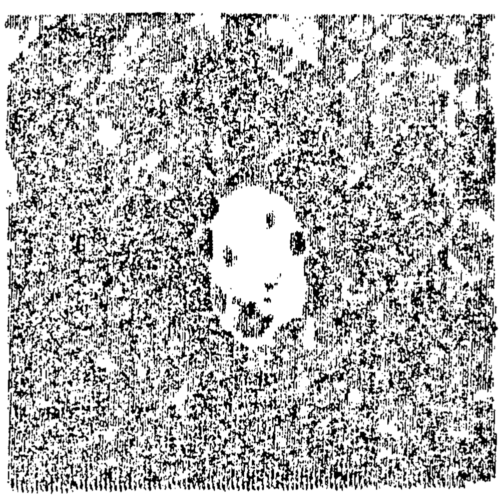
INSTRUMENTED IMPACT TEST SUMMARY

CLAMPED PLATE (3" x 3" test section)

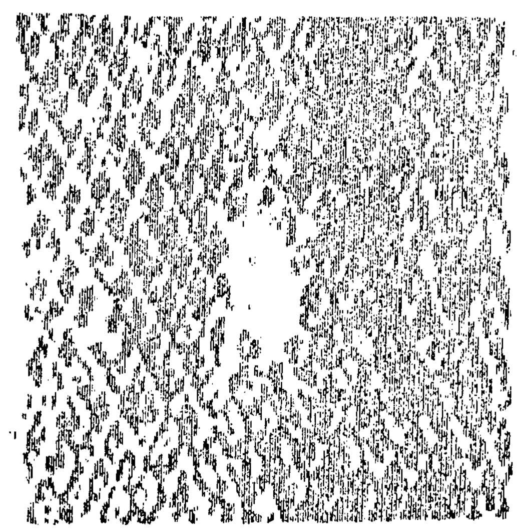
	24 PLY BASELINE		(1x1) MAGNAWEAVE		(1x1) 1/2 FIXED MAGNAWEAVE	
	MEAN	C.V.	MEAN	C.V.	MEAN	C.V.
THROUGH PENETRATION E (ft–lb)	55.0		44.3		29.1	
AT PEAK LOAD E (ft–lb)	13.4		10.3		6.0	
PEAK LOAD P (lbf)	1892		1403		806	
INITIAL DAMAGE P (lbf)	716	5.4%	621	16%	725	13%
INITIAL DAMAGE E (ft–lb)	1.7	16%	1.9	26%	4.3	32%

4 Ft–Lb ENERGY IMPACT TEST

1/2 inch radius 8.35lbm 5.5ft/sec



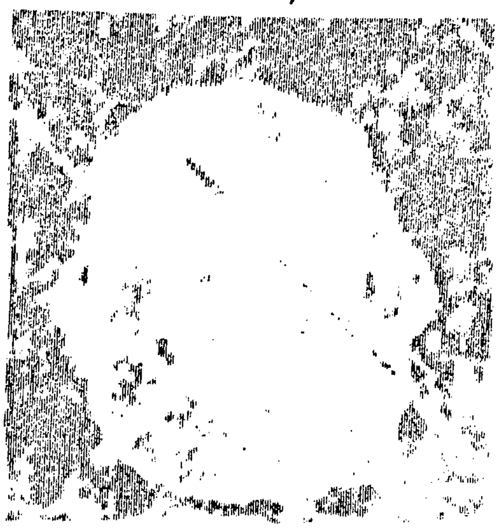
24 Ply Gr/Ep



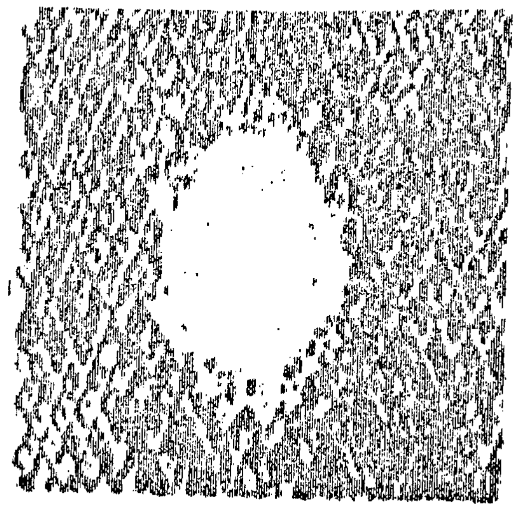
(1x1) MAGNAWEAVE

THROUGH PENETRATION IMPACT TEST

1/2 inch radius   32.7lbm   15ft/sec



24 Ply Gr/Ep
Eo= 55Ft–Lb



(1x1) MAGNAWEAVE
Eo= 44Ft–Lb

STATIC TEST SUMMARY

	C12000/3501 1x1 BRAID		24 Ply AS/3501 (42/50/8)		C12000/3501 (1x1) 1/2 FIXED	
	MEAN	C.V.	MEAN	C.V.	MEAN	C.V.
F_1^{tu}, ksi	96.8	9.3%	132.0	7.4%	108.7	6.1%
F_2^{tu}, ksi	5.0	10.0%	60.4	9.6%	3.3	19.5%
F_1^{cu}, ksi	62.1	14.5%	60.9	16.0%	68.6	17.6%
E_{11}^{t}, msi	13.1	19.5%	9.5	2.8%	15.4	12.3%
E_{22}^{t}, msi	1.5	9.7%	4.5	13.6%	1.4	9.7%
E_{11}^{c}, msi	11.0	21.8%	8.8	5.8%	13.5	19.8%
ν_{12}	1.06	51%	.42	–	.81	21.2%
ν_{21}	.067	6.7%	.225	2.8%	.04	44.9%
e_1^{tu}	.00773	13.8%	.01393	7.9%	.00733	10.8%
e_1^{cu}	.00640	10.2%	.00711	20.4%	.00533	15.7%
e_2^{tu}	.00324	9.7%	.01474	5.5%	.00249	21.3%

STATIC TEST SUMMARY (cont'd)

	C12000/3501 1x1 BRAID		24 Ply AS/3501 (42/50/8)		C12000/3501 (1x1) 1/2 FIXED	
	MEAN	C.V.	MEAN	C.V.	MEAN	C.V.
F_1^{tu}, ksi (D=.25) G	95.8	11.7%	64.5	2.3%	93.8	9.7%
F_1^{tu}, ksi (D=.25) N	127.8	11.7%	86.0	2.3%	125.1	9.2%
F_1^{cu}, ksi (D=.25) G	45.5	12.2%	58.4	6.1%	45.9	11.6%
F_1^{cu}, ksi (D=.25) N	60.6	12.2%	77.8	6.1%	61.2	11.6%
F_{br}^{c}, ksi (D=.25)	48.6	3.8%	83.7	9.5%	52.5	15.3%
F_{br}^{t}, ksi (D=.25) e/D = 2.5	26.5	6.7%	98.2	5.5%	41.0	21.0%

G = GROSS STRESS
N = NET STRESS

CONCLUSIONS

- BRAID LIMITS EXTENT OF IMPACT DAMAGE IN Gr/Ep BUT DOES NOT INCREASE DAMAGE THRESHOLD
- ELASTIC PROPERTIES SIMILAR TO COMPARABLE ANGLE–PLIED LAMINATES
- TRANSVERSE PROPERTIES POOR
- POISSON RATIO EXCESSIVE ($\nu_{xy} > .8$)

PROVISION MUST BE MADE TO "LAY IN" TRANSVERSE FIBERS (EST. 10% 90° TO GIVE $\nu_{xy} \leq .5$)

- BRAID INSENSITIVE TO 1/4" DIAMETER HOLE
- MANUFACTURING NOT STRAIGHTFORWARD

FRACTURE BEHAVIOR OF CERAMIC COMPOSITES

AFOSR CONTRACT NO.

F49620-82-C-0041

MECHANICS OF COMPOSITES REVIEW

OCTOBER 1983

OBJECTIVES

- o ATTEMPT TO IMPROVE STRENGTH AND FRACTURE TOUGHNESS OF A CERAMIC BY THE ADDITION OF REINFORCING WHISKERS
- o EXPERIMENTAL
  - . FABRICATE UNREINFORCED AND WHISKER REINFORCED CERAMICS
  - . EXPERIMENTALLY MEASURE FLEXURE STRENGTH AND FRACTURE TOUGHNESS
- o THEORETICAL
  - . ATTEMPT TO UNDERSTAND FAILURE BEHAVIOR BY COMPARING VARIOUS THEORIES TO EXPERIMENTAL RESULTS

CONCLUSIONS

- o EXPERIMENTAL
  - . MATERIALS ARE VERY BRITTLE. NO PLASTICITY OR STABLE CRACK GROWTH
  - . CRACKS GROW THROUGH THE MATRIX. WHISKERS APPEAR TO PULL OUT
  - . INCREASING WHISKER CONTENT LEADS TO INCREASED STRENGTH AND TOUGHNESS
  - . POOR PROPERTIES OF UNREINFORCED MATERIAL ARE ATTRIBUTED TO LARGE GRAIN SIZE
- o THEORETICAL
  - . DAMAGE ZONE CONCEPT CAN NOT BE REALISTICALLY APPLIED TO FRACTURE TOUGHNESS OF COMPOSITES
  - . STRENGTH OF COMPOSITES MAY BE RELATED TO INHERENT CRACK LENGTH WITHIN MATRIX. WHISKERS MAY ACT AS EITHER GRAIN GROWTH INHIBITORS OR CRACK ARRESTORS

EXPERIMENTAL MATERIALS

CONSTITUENTS

Material	Modulus	Density	Diameter	Aspect Ratio
—	Msi	gm/cc	μm	—
SiC Whiskers	63.5	3.21	0.3-1.3	8-130

Material	Grain Size	Modulus	Density	K_{IC}
—	μm	Msi	gm/cc	ksi$\sqrt{in}$
Al_2O_3 Matrix	2-50	32.1-57.0	3.4-3.99	2.71-4.01

FABRICATION

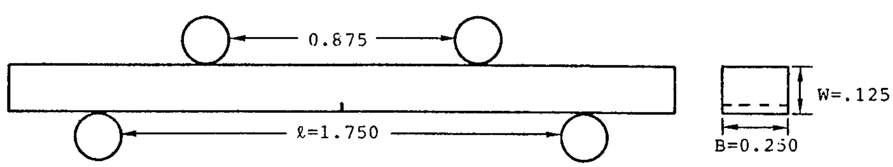
- o Powder-Whisker Blends
- o Hot Pressed at 4000 psi
- o Heated above 1750°C for 90 minutes
- o Maximum Temperature 1860°C for 15 minutes

COMPOSITES

Unreinforced Al_2O_3 10 v/o SiC/Al_2O_3 20 v/o SiC/Al_2O_3

EXPERIMENTAL TESTS

Four Point Bending of Single Edge Notched Beam



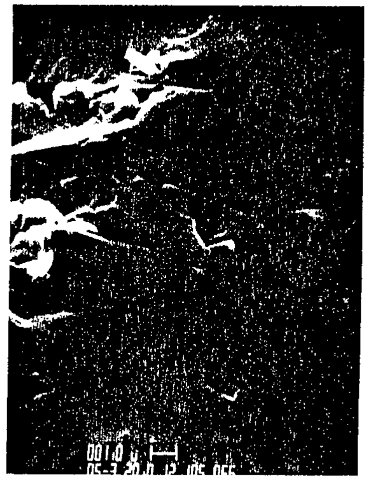
TEST MATRIX

Material	SiC Content	Flexure*	Flexure*	Flexure*	Flexure*	SEM
-	v/o	$\frac{a}{W}$ = 0.0	$\frac{a}{W}$ = 0.25	$\frac{a}{W}$ = 0.35	$\frac{a}{W}$ = 0.50	-
Al_2O_3	0	Spec. A1-A10	D1-D5	G1-G5	J1-J5	8
10 SiC/Al_2O_3	10	Spec. B1-B5	E1-E5	I1-I5	L1-L5	8
20 SiC/Al_2O_3	20	Spec. C1-C5	F1-F5	H1-H5	K1-K5	8

Total of 65 mechanical tests

*Note: a/W refers to initial notch depth

SCANNING ELECTRON MICROGRAPHS

100X	5000X	5000X
		
Unreinforced Al_2O_3	10 v/o SiC/Al_2O_3	20 v/o SiC/Al_2O_3

EXPERIMENTAL RESULTS

$$\sigma = \frac{3P_{max}L}{4BW^2} \qquad K_{IC} = \frac{3P_{max}L\sqrt{a}}{4BW^2}\left[1.992-2.468\left(\frac{a}{W}\right)+12.97\left(\frac{a}{W}\right)^2-23.17\left(\frac{a}{W}\right)^3+24.80\left(\frac{a}{W}\right)^4\right]$$

	Unreinforced Al_2O_3				10 v/o SiC/Al_2O_3				20 v/o SiC/Al_2O_3			
Notch Depth	Maximum Load	Avg. Load	Avg. σ	Avg. K_{IC}	Maximum Load	Avg. Load	Avg. σ	Avg. K_{IC}	Maximum Load	Avg. Load	Avg. σ	Avg. K_{IC}
a/W	lbs.	lbs.	ksi	ksi$\sqrt{in}$	lbs.	lbs.	ksi	ksi$\sqrt{in}$	lbs.	lbs.	ksi	ksi$\sqrt{in}$
0.0	8.2-12.8	10.3	3.45	—	62.5-131	104	34.9	—	121-173	150	50.3	—
0.25	7.5- 9.3	8.10	2.72	0.93	30.9-34.1	32.1	10.8	3.67	39.2-44.5	42.6	14.3	4.86
0.35	4.3- 6.0	5.0	1.68	0.74	18.2-19.5	18.9	6.35	2.79	22.3-25.8	24.0	3.06	3.54
0.50	2.5- 3.3	2.93	9.85	0.65	12.2-13.1	12.6	4.23	2.82	15.3-16.3	15.8	5.31	3.52
	STRENGTH = 3.45 ksi; K_{IC} = 0.77 ksi$\sqrt{in}$				STRENGTH = 34.9 ksi; K_{IC} = 3.09 ksi$\sqrt{in}$				STRENGTH = 50.3 ksi; K_{IC} = 3.97 ksi$\sqrt{in}$			

CRACK TIP DAMAGE ZONE

HYPOTHESIS

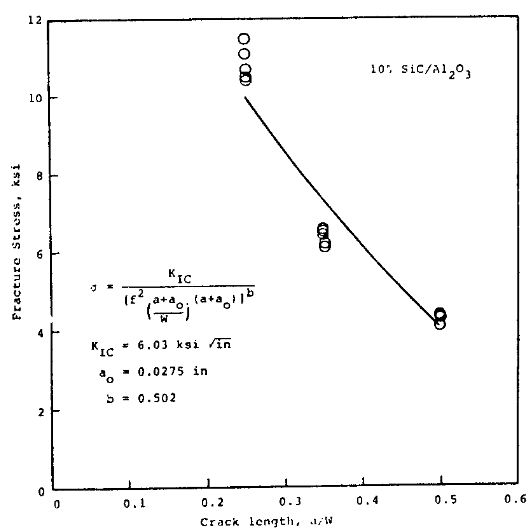
- APPLY LINEAR ELASTIC FRACTURE MECHANICS TO COMPOSITES BY ASSUMING A SMALL DAMAGE ZONE EXISTS AT CRACK TIP.
- "DAMAGE ZONE" CAN BE RELATED TO CONSTITUENTS AND VOLUME FRACTION

TEST

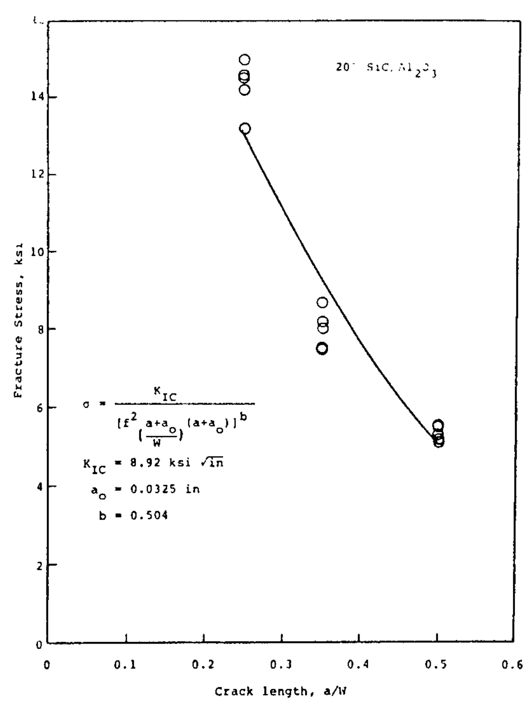
- ASSUME ACTUAL CRACK LENGTH IS $a+a_o$. a_o- DAMAGE ZONE SIZE
- FIT CURVES TO σ VS. a DATA FOR DIFFERENT COMPOSITES
- $\sigma = K_{IC}/\{(a+a_o)^{\frac{1}{2}}[1.992-2.468(\frac{a+a_o}{W})+12.97(\frac{a+a_o}{W})^2-23.17(\frac{a+a_o}{W})^3+24.80(\frac{a+a_o}{W})^4]\}$
- CURVE FIT SHOULD RESULT IN
  - NEARLY SAME K_{IC} FOR 10% AND 20% COMPOSITES
  - DAMAGE ZONE SIZE ON THE ORDER OF COMPOSITE MICROSTRUCTURE
  - DAMAGE ZONE SIZE WHICH DECREASES AS VOLUME FRACTION INCREASES

DAMAGE ZONE - RESULTS

10 v/o SiC/Al_2O_3



20 v/o SiC/Al_2O_3



- DAMAGE ZONE SIZE REQUIRED TO FIT DATA SEEMS UNREALISTICALLY LARGE

INHERENT CRACK LENGTH

- HYPOTHESIS
  - STRENGTH OF CERAMIC COMPOSITES IS GOVERNED BY INHERENT FLAWS IN MATRIX
  - WHISKERS LIMIT FLAW SIZE TO A LENGTH ON THE ORDER OF THE MEAN FREE PATH THROUGH THE MATRIX
- TEST
  - COMPUTE MEAN FREE PATH AS A FUNCTION OF WHISKER SHAPE AND VOLUME FRACTION
  - TREAT MEAN FREE PATH AS INHERENT CRACK LENGTH AND USE FRACTURE MECHANICS TO PREDICT CRITICAL STRESS
  - COMPARE PREDICTED STRESSES TO MEASURED FLEXURAL STRENGTHS

INHERENT CRACK LENGTH

INHERENT CRACK LENGTH

MEAN FREE PATH

$$1 = \frac{4}{3}\, a \left(\frac{b}{a}\right)^{2/3} \left(\frac{1}{V_r} - 1\right)$$

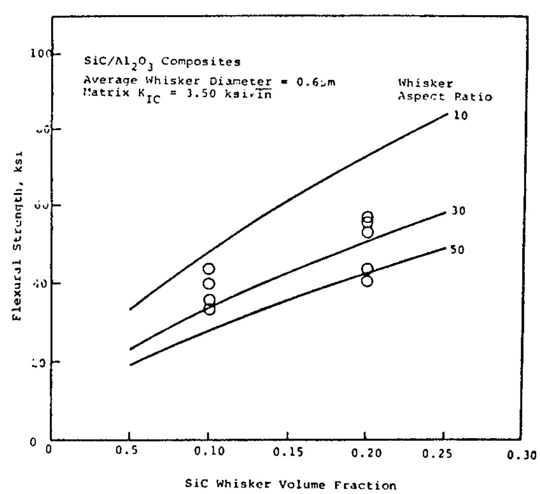
a - whisker diameter

$\frac{b}{a}$ - whisker aspect ratio

V_r - whisker volume fraction

CRITICAL STRESS

$$\sigma = \frac{K_{IC}}{f\left(\frac{\ell}{W}\right)\sqrt{\pi \ell}}$$



ANALYTICAL RESULTS FOR POSTBUCKLING BEHAVIOR OF ORTHOTROPIC COMPOSITE PLATES IN COMPRESSION AND IN SHEAR

BY

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NINTH ANNUAL MECHANICS OF COMPOSITES REVIEW

DAYTON, OH

OCTOBER 24-26, 1983

RESEARCH OBJECTIVES

- AN UNDERSTANDING OF THE POSTBUCKLING BEHAVIOR OF COMPOSITE PLATES IS NEEDED

- THIS PAPER PRESENTS POSTBUCKLING RESULTS FOR LONG COMPOSITE PLATES WITH CONSTRAINTS AT THE PLATE EDGES THAT REPRESENT AN UPPER LIMIT TO THE CONSTRAINTS EXPECTED IN ACTUAL STRUCTURES AND EXPERIMENT

    - CONSTRAINTS CONSIDERED RESTRICT TRANSVERSE INPLANE DISPLACEMENT (v)

        - FOR COMPRESSION AND SHEAR LOADING THE LONG EDGES ARE NOT FREE TO MOVE TRANSVERSELY ($v = 0$ AT THE EDGES $y = 0, b$)

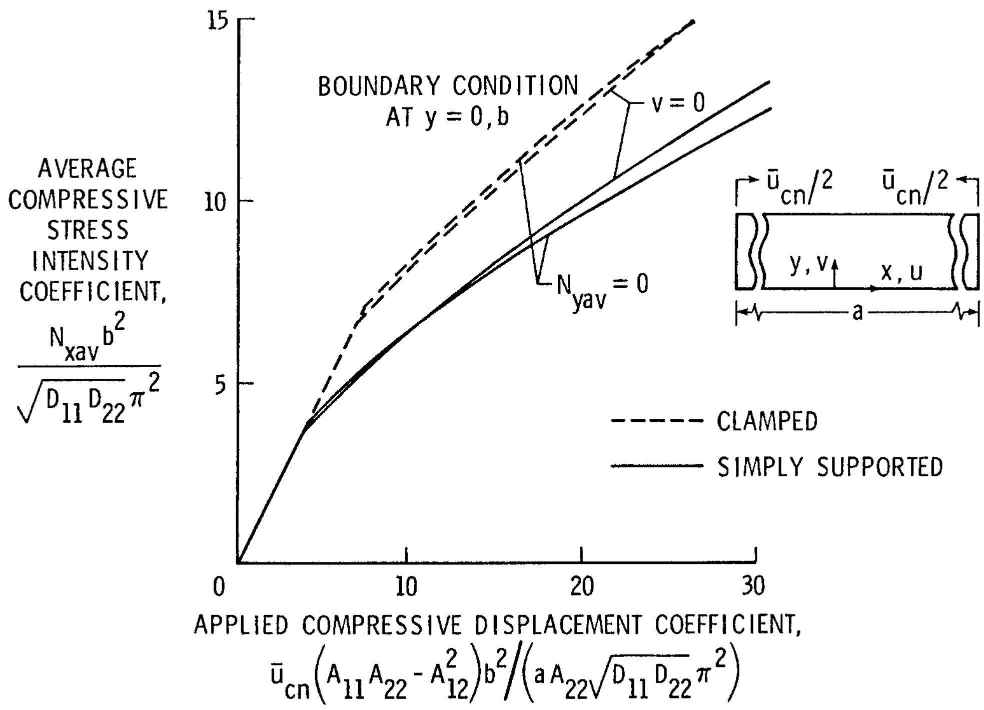
        - OR FOR SHEAR LOADING THE LONG EDGES MOVE IN AS THEY WOULD FOR A PANEL IN A RIGID FRAME PINNED AT THE CORNERS

SUMMARY

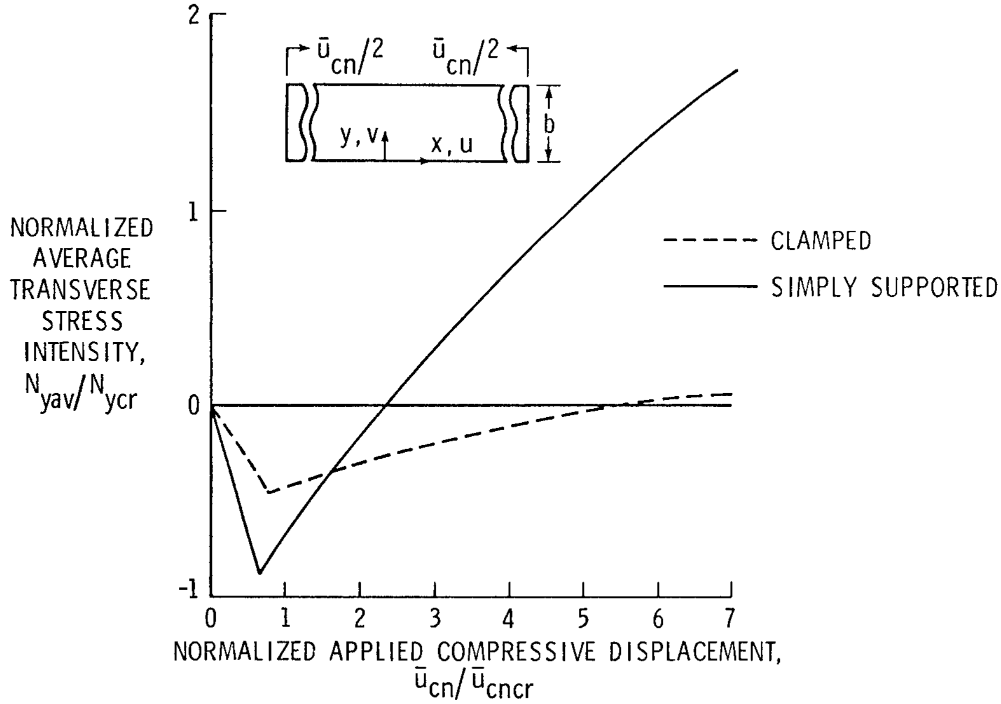
RESULTS FOR LONG PLATES WITH TRANSVERSE INPLANE DISPLACEMENTS (v) CONSTRAINED AT THE EDGES ARE COMPARED TO RESULTS FOR LONG PLATES WITH THE TRANSVERSE STRESS INTENSITY $\left(N_y\right)$ ZERO ON THE AVERAGE AT THE EDGES

- COMPRESSION POSTBUCKLING RESULTS ARE INSENSITIVE TO WHETHER $v = 0$ OR $N_{yav} = 0$ AT THE EDGES
- SHEAR POSTBUCKLING RESULTS ARE SENSITIVE TO INPLANE v CONSTRAINTS
  - SHEAR STIFFNESS IS MUCH LARGER WITH CONSTRAINTS THAN WITHOUT CONSTRAINTS
  - TRANSVERSE STRESS IS LARGE WITH CONSTRAINTS (ZERO WITHOUT)
  - LONGITUDINAL STRESS IS ABOUT THE SAME FOR ISOTROPIC PLATES WITH OR WITHOUT CONSTRAINTS, BUT LONGITUDINAL STRESS IS LARGER FOR THE ±45° LAMINATED PLATE WITH CONSTRAINTS THAN WITHOUT CONSTRAINTS

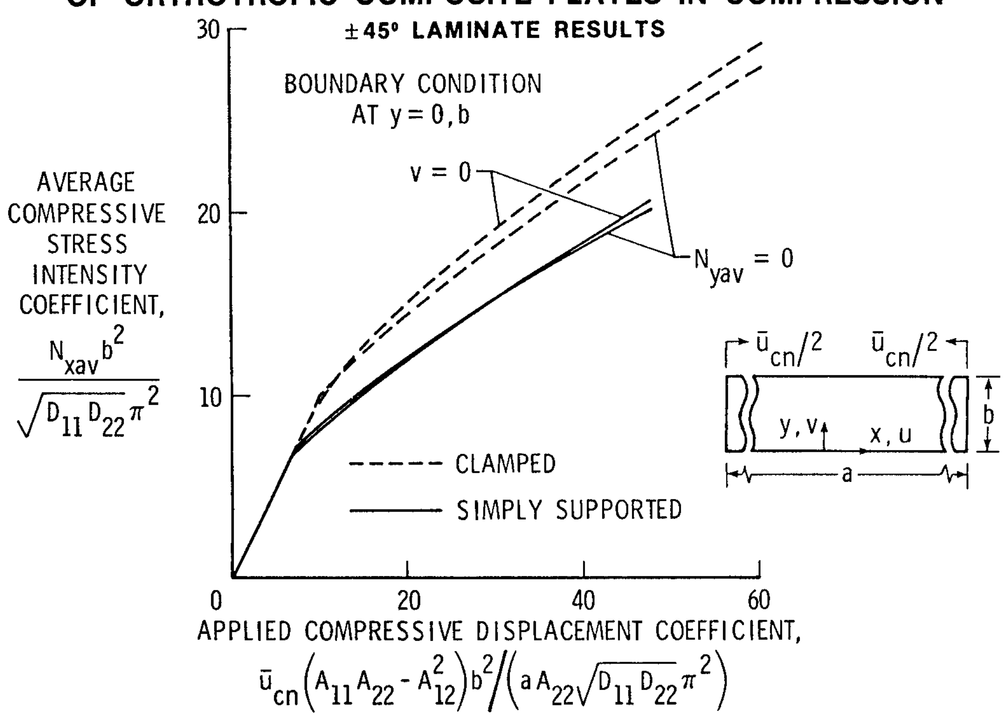
CHARACTERISTIC CURVES FOR POSTBUCKLING BEHAVIOR OF ISOTROPIC PLATES IN COMPRESSION



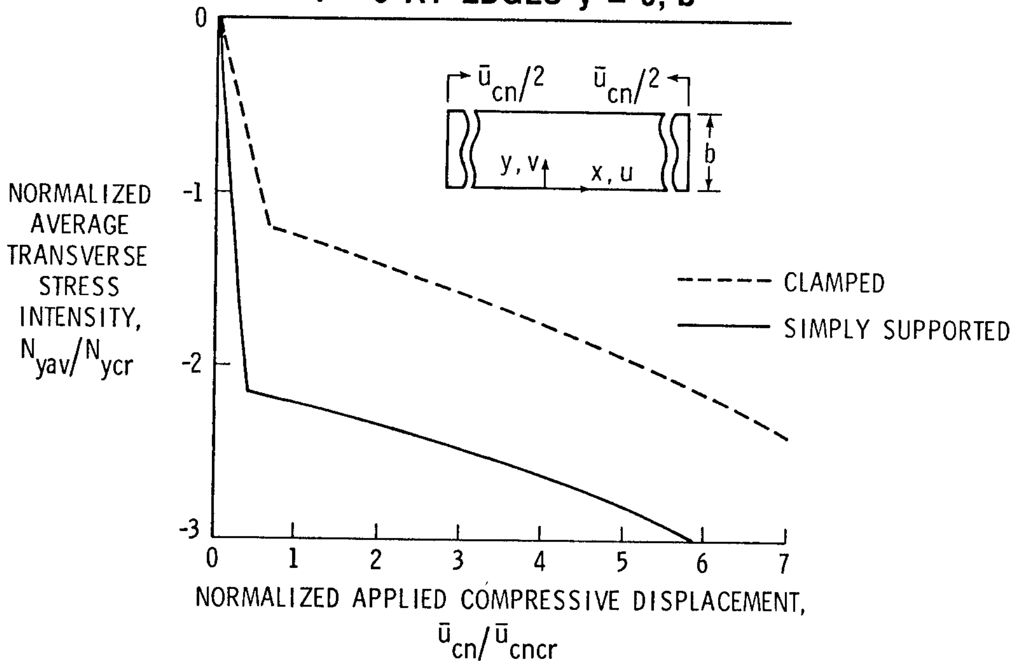
N_{yav} FOR ISOTROPIC PLATES IN COMPRESSION WITH INPLANE CONDITION v = 0 AT EDGES y = 0, b



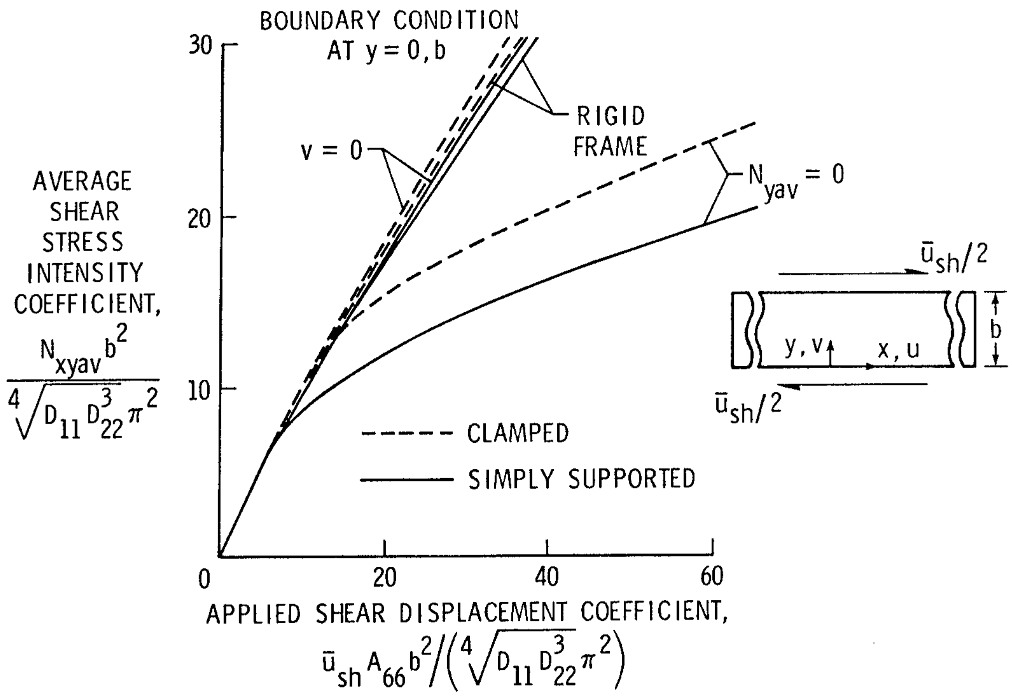
CHARACTERISTIC CURVES FOR POSTBUCKLING BEHAVIOR OF ORTHOTROPIC COMPOSITE PLATES IN COMPRESSION

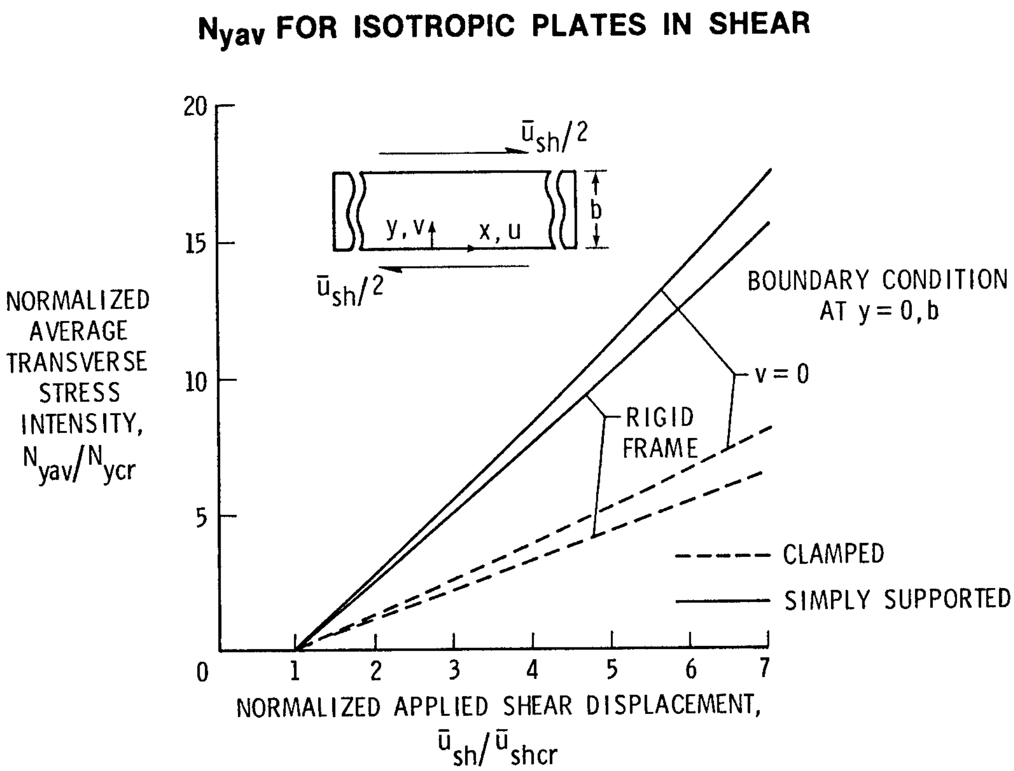


N_{yav} FOR ±45° LAMINATED COMPOSITE PLATE IN COMPRESSION WITH INPLANE CONDITION v = 0 AT EDGES y = 0, b

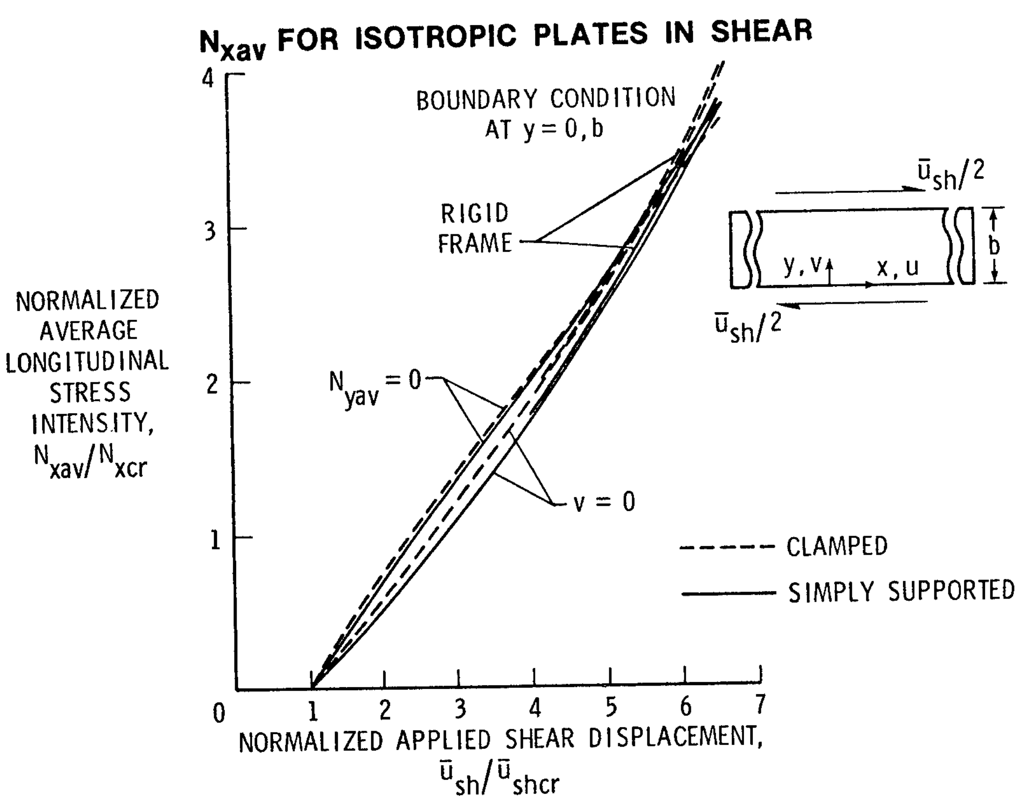


CHARACTERISTIC CURVES FOR POSTBUCKLING OF ISOTROPIC PLATES IN SHEAR





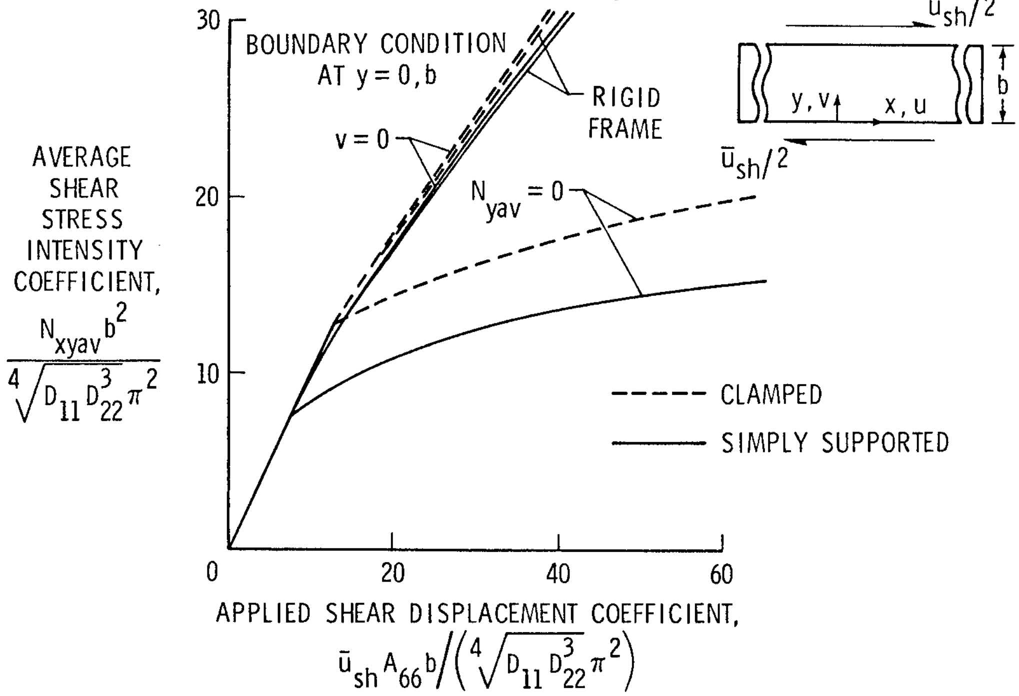
N_{yav} FOR ISOTROPIC PLATES IN SHEAR
20
15
10
5
0
1
2
3
4
5
6
7
$\bar{u}_{sh}/2$
y, v
x, u
b
NORMALIZED AVERAGE TRANSVERSE STRESS INTENSITY, N_{yav}/N_{ycr}
BOUNDARY CONDITION AT y = 0, b
v = 0
RIGID FRAME
CLAMPED
SIMPLY SUPPORTED
NORMALIZED APPLIED SHEAR DISPLACEMENT, $\bar{u}_{sh}/\bar{u}_{shcr}$



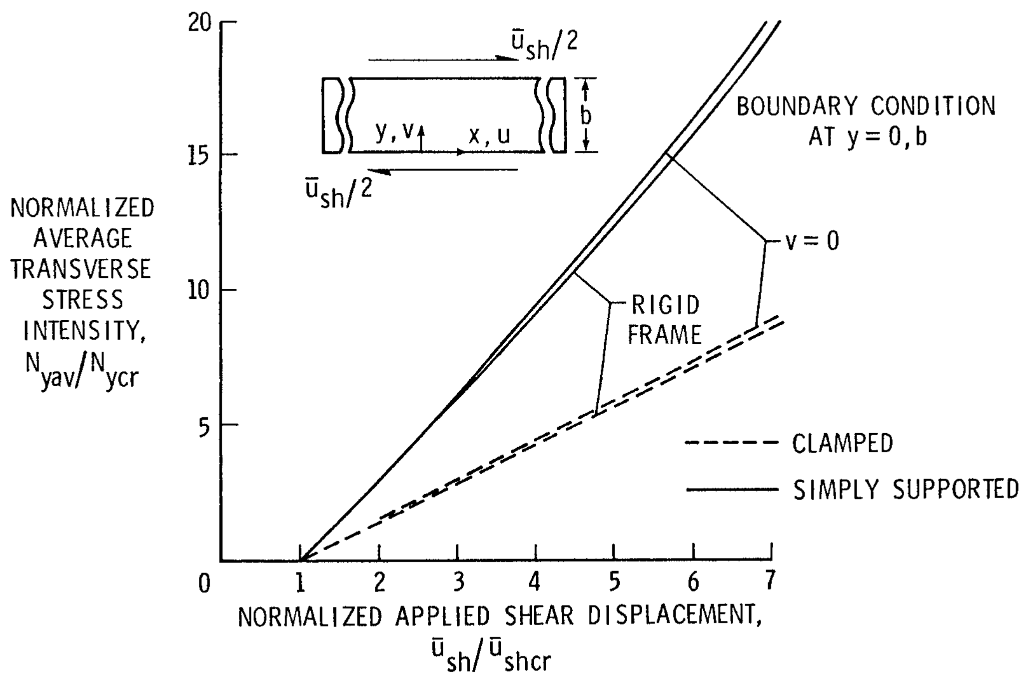
N_{xav} FOR ISOTROPIC PLATES IN SHEAR
4
3
2
1
0
1
2
3
4
5
6
7
BOUNDARY CONDITION AT y = 0, b
RIGID FRAME
$N_{yav} = 0$
v = 0
$\bar{u}_{sh}/2$
y, v
x, u
b
NORMALIZED AVERAGE LONGITUDINAL STRESS INTENSITY, N_{xav}/N_{xcr}
CLAMPED
SIMPLY SUPPORTED
NORMALIZED APPLIED SHEAR DISPLACEMENT, $\bar{u}_{sh}/\bar{u}_{shcr}$

CHARACTERISTIC CURVES FOR POSTBUCKLING BEHAVIOR OF ORTHOTROPIC COMPOSITE PLATES IN SHEAR

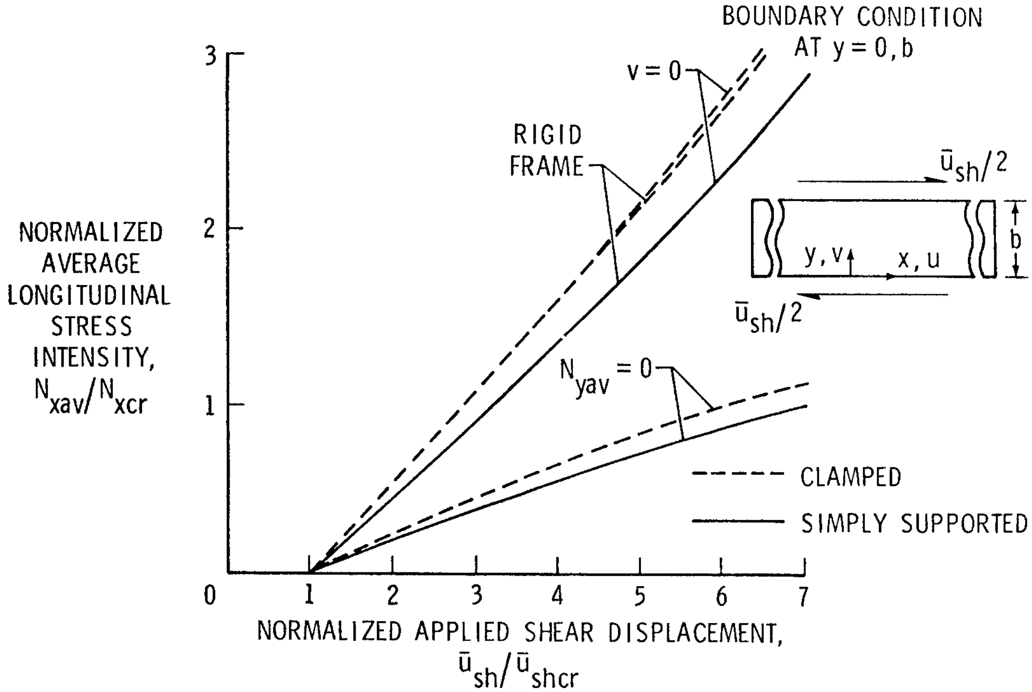
±45° LAMINATE RESULTS



N_{yav} FOR ±45° LAMINATED COMPOSITE PLATE IN SHEAR



N_{xav} FOR ±45° LAMINATED COMPOSITE PLATE IN SHEAR



EXPERIMENTAL AND ANALYTICAL EVALUATION OF NONLINEAR MECHANICAL RESPONSE IN NOTCHED LAMINATES

by

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OBJECTIVES

- ESTABLISH EXPERIMENTALLY THE ROLE OF NONLINEAR RESPONSE IN 0/90, ± 45 AND OTHER LAMINATE STACKING SEQUENCES WHICH ARE PRONE TO GIVE RISE TO NONLINEAR RESPONSE, ON THE BEHAVIOR OF PIN LOADED AND NOTCHED LAMINATES.
- DEVELOP FINITE ELEMENT APPROACHES CAPABLE OF MODELLING TYPICAL NONLINEAR RESPONSE OF LAMINATES AND DETERMINE EXPERIMENTAL RESULTS REFLECTING THE INFLUENCE OF NONLINEARITY CAN BE REPRODUCED ANALYTICALLY.
- DEVELOP IMPROV ED STRESS-STRAIN LAWSFOR INCLUSION IN FINITE ELEMENT SOLUTIONS WHERE NEEDED. ESTABLISH WHERE TWO-DIMENSIONAL APPROACHES ARE INADEQUATE.
- ESTABLISH THE EXTENT TO HWICH NONLINEAR RESPONSE INFLUENCES FAILURE IN NOTCHED LAMINATES.

CONCLUSIONS TO DATE

- EXPERIMENTAL RESULTS BASED ON THE APPLICATION OF THE MOIRE METHOD HAVE ESTABLISHED THE SIGNIFICANT DEGREE TO WHICH NONLINEAR RESPONSE EFFECTS FAILURE OF PIN OR BOLTED JOINTS, ESPECIALLY IN 0/90 OR ± 45 LAMINATES.
- 0/90 LAMINATESARE NONLINEAR PRIMARILY IN SHEAR, AND EFFECT THE RESPONSE OF PIN LOADED LAMINATES PRIMARILY IN THE REGION IN FRONT OF THE PIN. ± 45 LAMINATE ARE NONLINEAR IN TENSION AND EFFECT PRIMARILY THE NET-SECTION REGION.
- ANALYTICAL RESULTS TO DATE FROM NONLINEAR ORTHOTROPIC FINITE ELEMENT APPROACHES GIVE QUALITATIVE AGGREEMENT WITH EXPERIMENTAL RESULTS, BUT FURTHER EFFORT ON THE ASSUMED STRESS-STRAIN LAW IS NEEDED TO GET SATISFACTORY QUANTITATIVE AGGREEMENT, FOR 0/90 LAMINATES. ± 45 LAMINATES HAVE NOT BEEN TREATED ANALYTICALLY AT THIS POINT.
- THREE DIMENSIONAL APPROACHES MAY BE NEEDED TO TREAT SOME ASPECTS OF THE PROBLEM, IE., INCIPIENT BEARING FAILURE

FORMAT OF PRESENTATION

- REVIEW OBSERVATIONS FROM PREVIOUS EXPERIMENTAL RESULTS*
- ANALYTICAL APPROACH
- PRELIMINARY ANALYTICAL RESULTS
- FUTURE EFFORTS

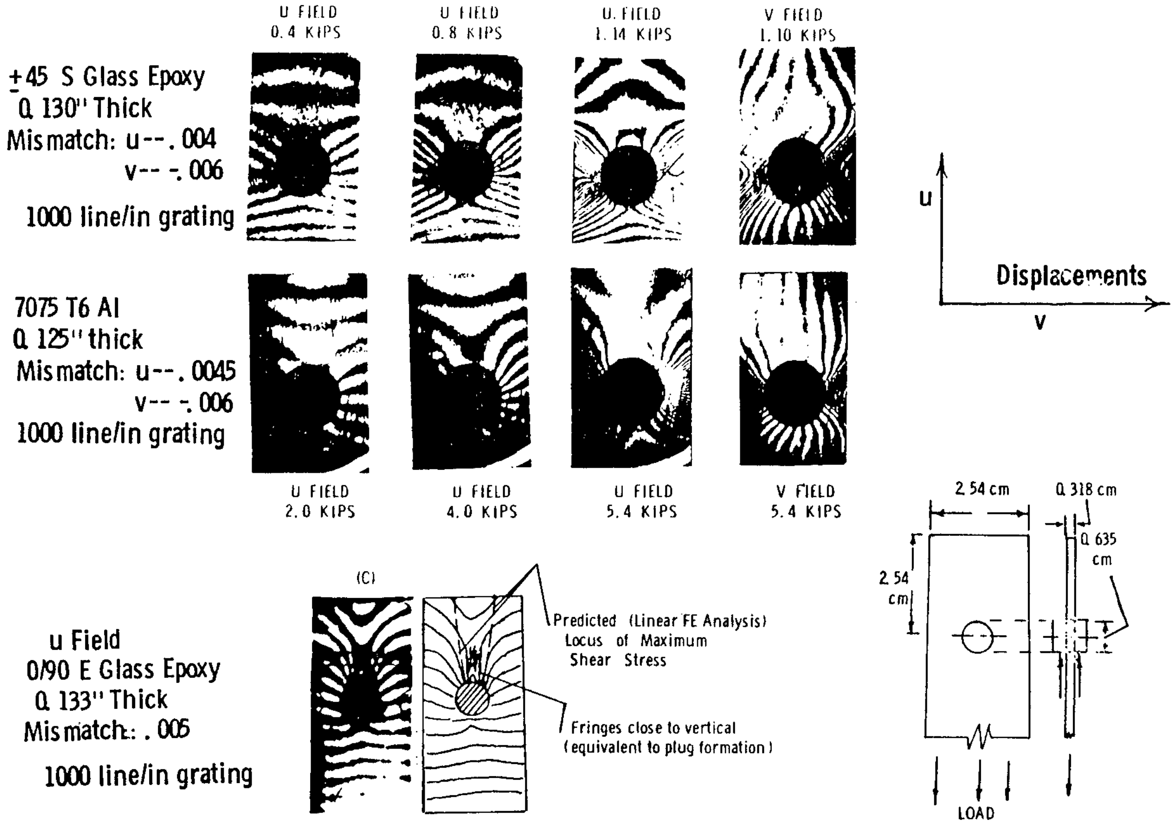
*1. Oplinger D. W. FIBROUS COMPOSITES IN STRUCTURAL DESIGN PP 575-602 Plenum Press (Dec. 1979)

2. Oplinger D. W. "Applications of MOIRE Methods to Evaluation of Structural Performance of Composite Materials" OPTICAL ENGINEERING. v. 21 pp. 626-632 (July 1982)

TEAM MAKEUP

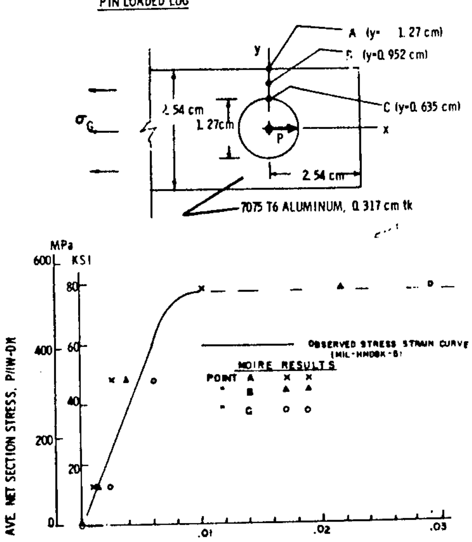
- D. Oplinger, analytical efforts
- C. E. Freese, FE program development
- K. R. Gandhi, nonlinear laminate analysis
- B. S. Parker, moire results
- S. Serabian, moire results

MOIRE FRINGE PATTERNS IN METAL AND COMPOSITE PIN LOADED LUGS

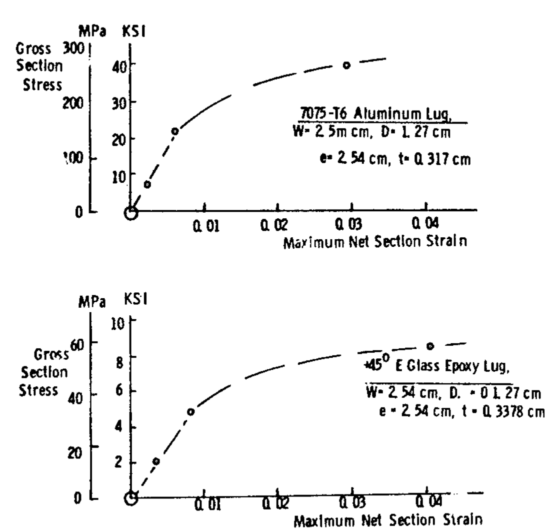


STRAIN MEASUREMENTS FROM MOIRE FRINGE PATTERNS, ALUMINUM AND ±45 GLASS EPOXY LUGS

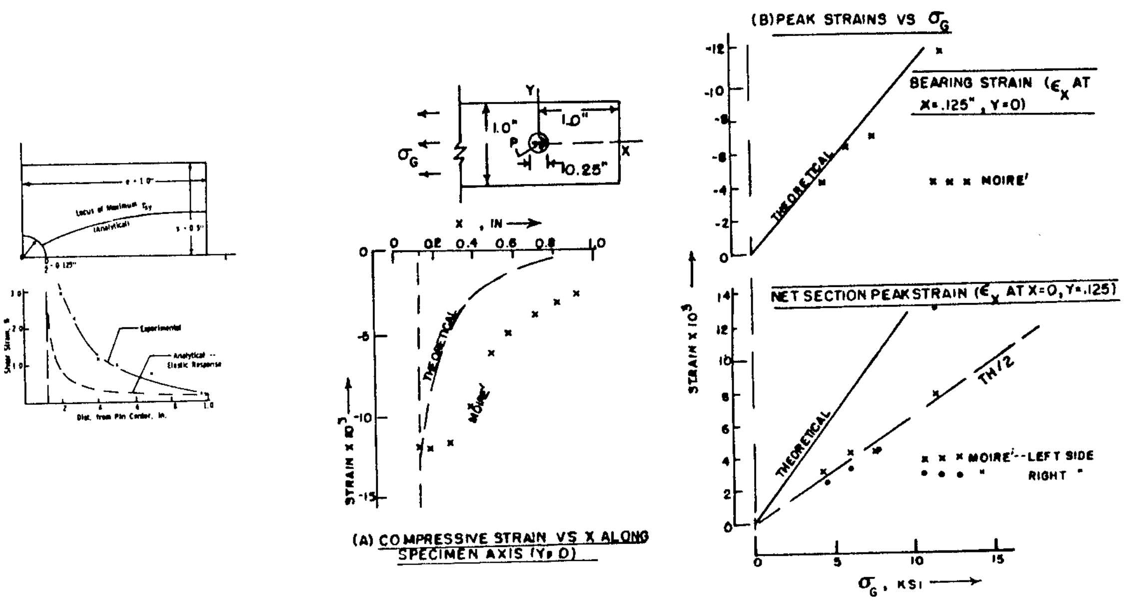
(A) DISTRIBUTION OF STRAINS IN NET SECTION OF ALUMINUM PIN LOADED LUG



(B) COMPARISON OF STRAIN FIELDS IN ALUMINUM AND DUCTILE COMPOSITE LUGS UNDER PIN LOAD



STRAIN MEASUREMENTS FROM MOIRE FRINGE PATTERNS, 0/90 GLASS EPOXY LUG



DIRECT METHOD OF NONLINEAR LAMINATE ANALYSIS

NONLINEAR LAMINATE ANALYSIS

1. GLOBAL LAMINATE STRAIN VECTOR $\underset{\sim}{\epsilon}^{xy} = \begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \gamma_{xy} \end{bmatrix}$

2. TRANSFORM ϵ^{xy} TO LAYER i MATERIAL AXES:

$$\begin{bmatrix} \epsilon_L \\ \epsilon_T \\ \gamma_{LT} \end{bmatrix}_i \equiv \underset{\sim}{\epsilon}_i^{LT} = [R]^{-1}[T]_i[R]\,\underset{\sim}{\epsilon}^{xy}$$

$$[R] = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 2 \end{bmatrix}; \quad [T]_i = \begin{bmatrix} c^2 & s^2 & 2sc \\ s^2 & c^2 & -2sc \\ -sc & sc & c^2-s^2 \end{bmatrix}$$

$c = \cos\theta_i$; $s = \sin\theta_i$

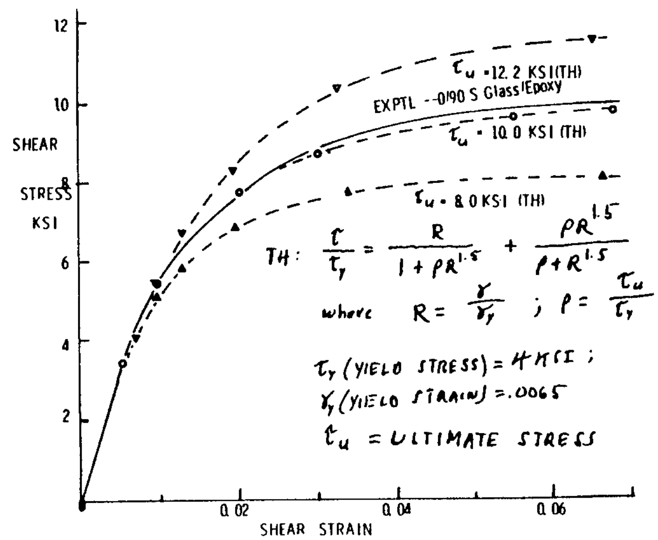
3. LAYER STRESS VECTOR $\underset{\sim}{\sigma}_i^{LT}$

$$\begin{bmatrix} \sigma_L \\ \sigma_T \\ \tau_{LT} \end{bmatrix}_i \equiv \underset{\sim}{\sigma}_i^{LT}; \quad \begin{aligned} \sigma_L &= C_{11}^i \epsilon_L + C_{12}^i \epsilon_T \\ \sigma_T &= C_{12}^i \epsilon_L + C_{22}^i \epsilon_T \end{aligned}$$

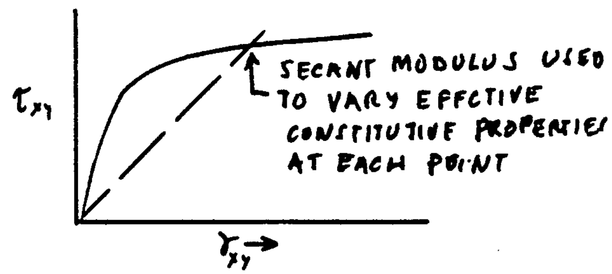
(NONLINEAR EFFECT) $\boxed{\tau_{LT} = H(\gamma_{LT})}$

4. GLOBAL STRESS VECTOR $\underset{\sim}{\sigma}^{xy}$

$$\underset{\sim}{\sigma}^{xy} = \frac{\sum_i [T]_i^{-1}\, t_i\, \underset{\sim}{\sigma}_i^{LT}}{\sum_i t_i}; \quad t_i = \text{l'yr thickness}$$

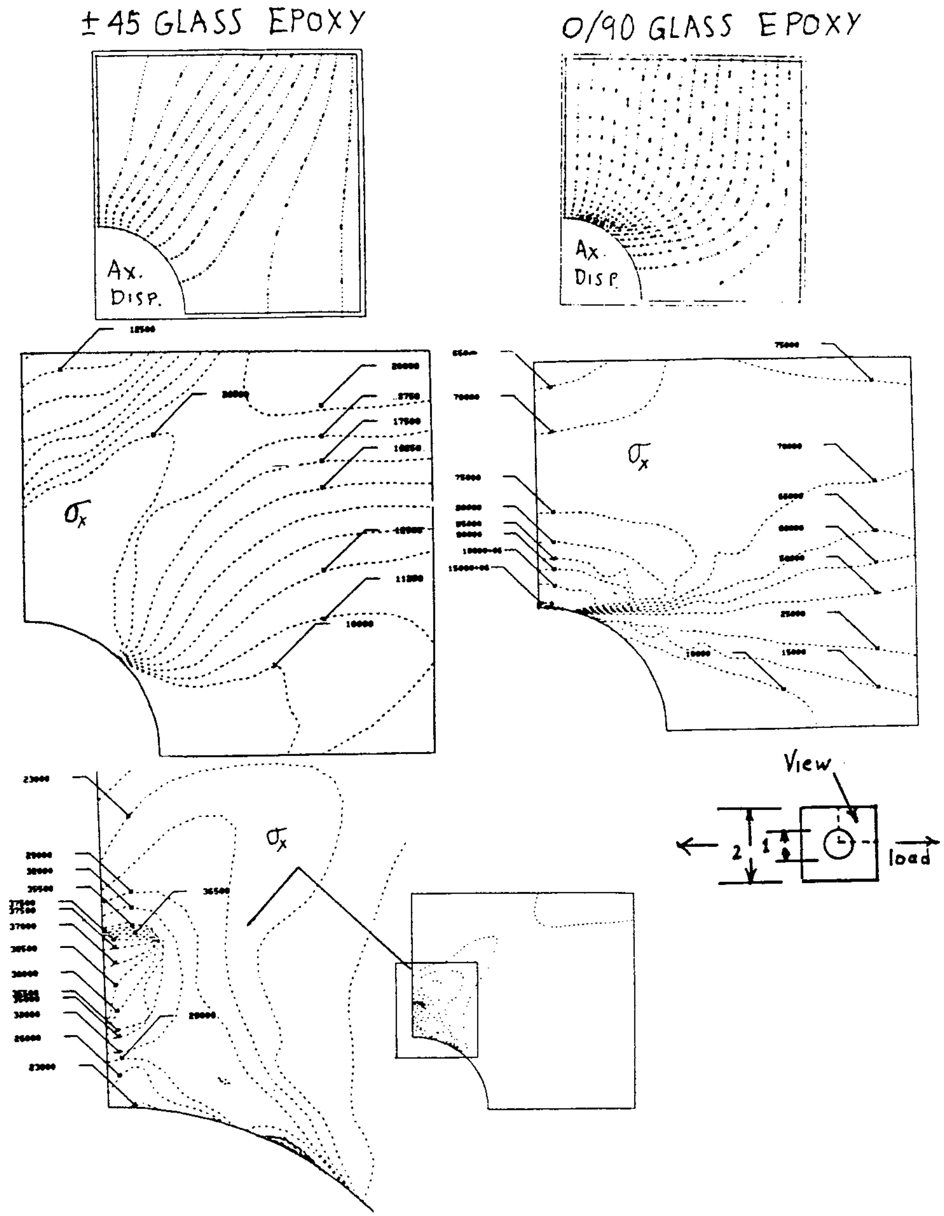


NONLINEAR ORTHOTROPIC FINITE ELEMENT APPROACH

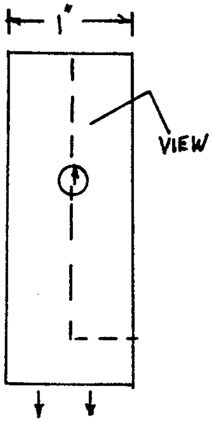


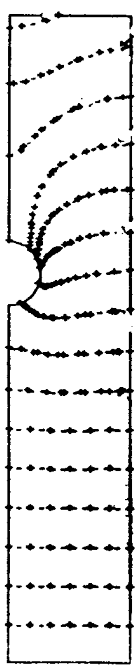
1. Get linear solution and strain at each GAUSS POINT, assuming initial shear modulus.
2. Feed strain values into NL laminate analysis to get stress vector at each point.
3. Divide stresses by strains to get effective new values of modulii for next iteration of FE analysis.
4. Repeat steps 2 and 3. until convergence indicated by no further changes in results.

ORTHOTROPIC NONLINEAR FINITE ELEMENT RESULTS FOR PLATE UNDER TENSION, UNLOADED HOLE

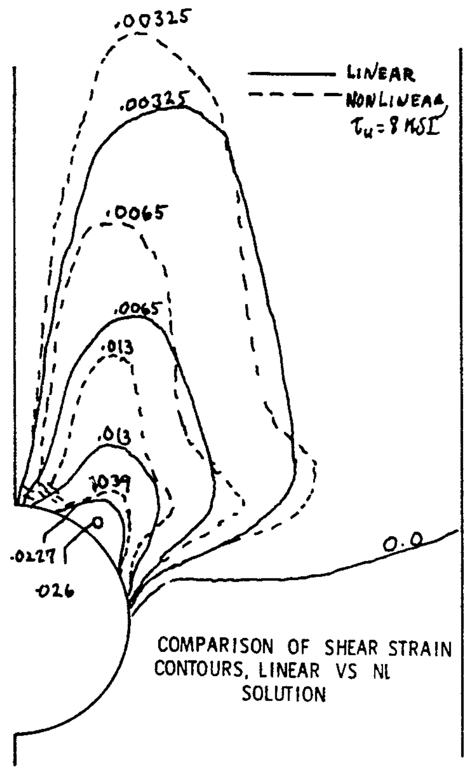


ORTHOTROPIC NONLINEAR FINITE ELEMENT RESULTS FOR PIN LOADED 0/90 GLASS EPOXY PLATE





COMPUTER GENERATED MOIRE FRINGES (AXIAL DISPLACEMENT CONTOURS)
($\tau_u = 8$ KSI)



COMPARISON OF SHEAR STRAIN CONTOURS, LINEAR VS NL SOLUTION

FUTURE EFFORTS

- RE-DO MOIRE RESULTS AND LOOK IN MORE DETAIL AT BEHAVIOR NEAR INCIPIENT FAILURE
- CONTINUE ANALYTICAL EFFORTS. DETERMINED WHERE MODIFIED STRESS-STRAIN LAW NEEDED DETERMINE WHERE THREE DIMENSIONAL APPROACHES NEEDED.
- CONDUCT 3D ANALYSES AS NEEDED.

BUCKLING OF SURFACE DELAMINATIONS IN A QUASI-ISOTROPIC LAMINATE

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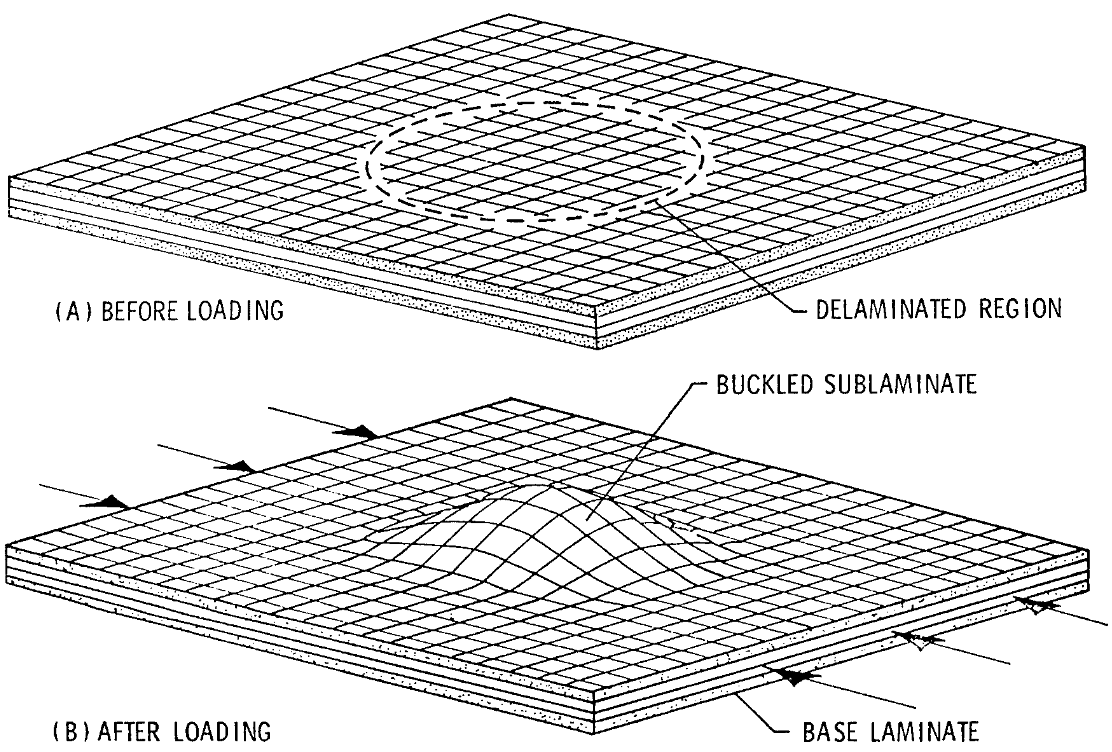
LOCAL BUCKLING OF A DELAMINATED LAMINATE



OBJECTIVES

- PREDICT THE BUCKLING OF A SUBLAMINATE IN A QUASI-ISOTROPIC LAMINATE
- DETERMINE THE EFFECT OF SUBLAMINATE PROPERTIES ON BUCKLING STRAIN

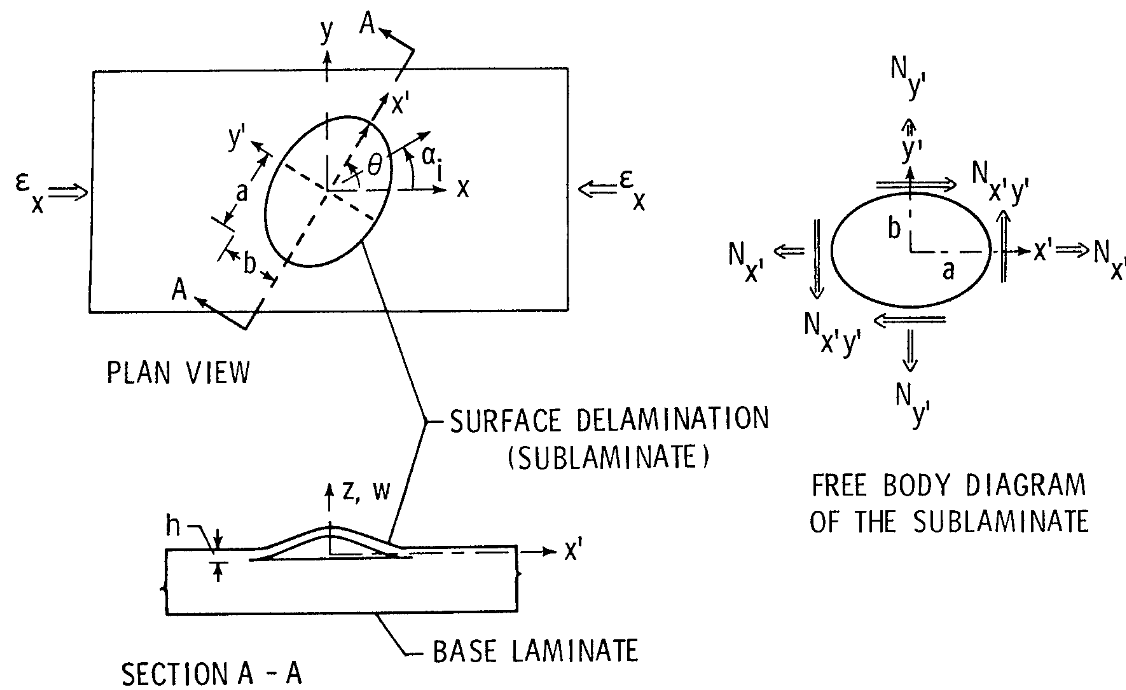
  FIBER ORIENTATION

  STACKING SEQUENCE

  SHAPE

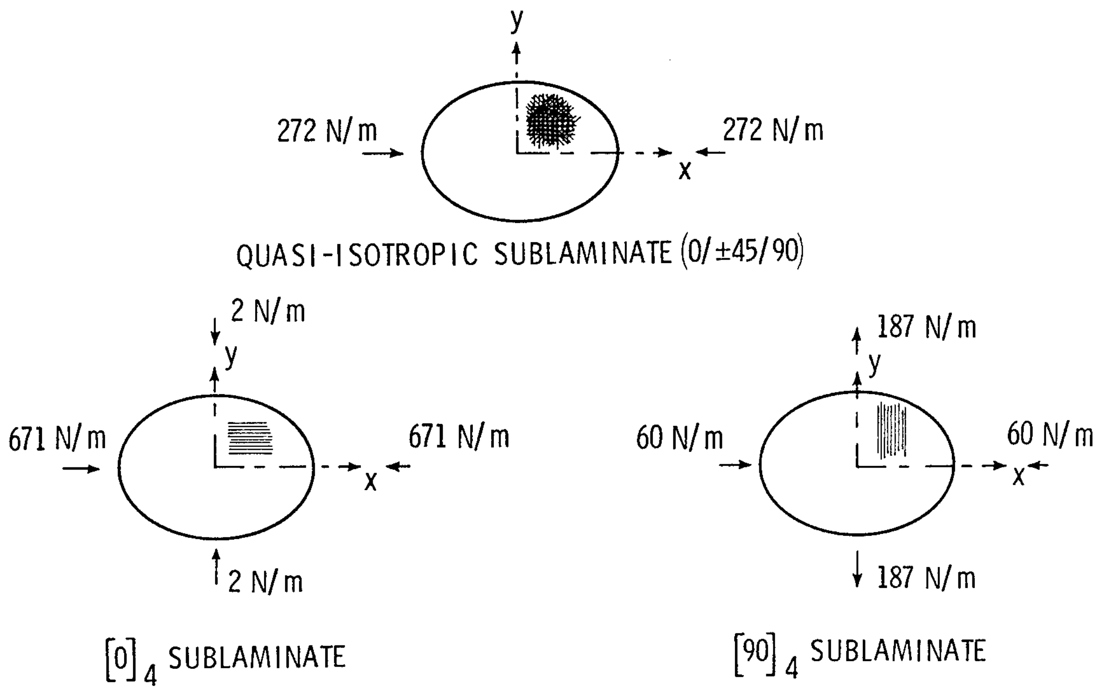
  SUBLAMINATE ORIENTATION

DESCRIPTION OF THE PROBLEM

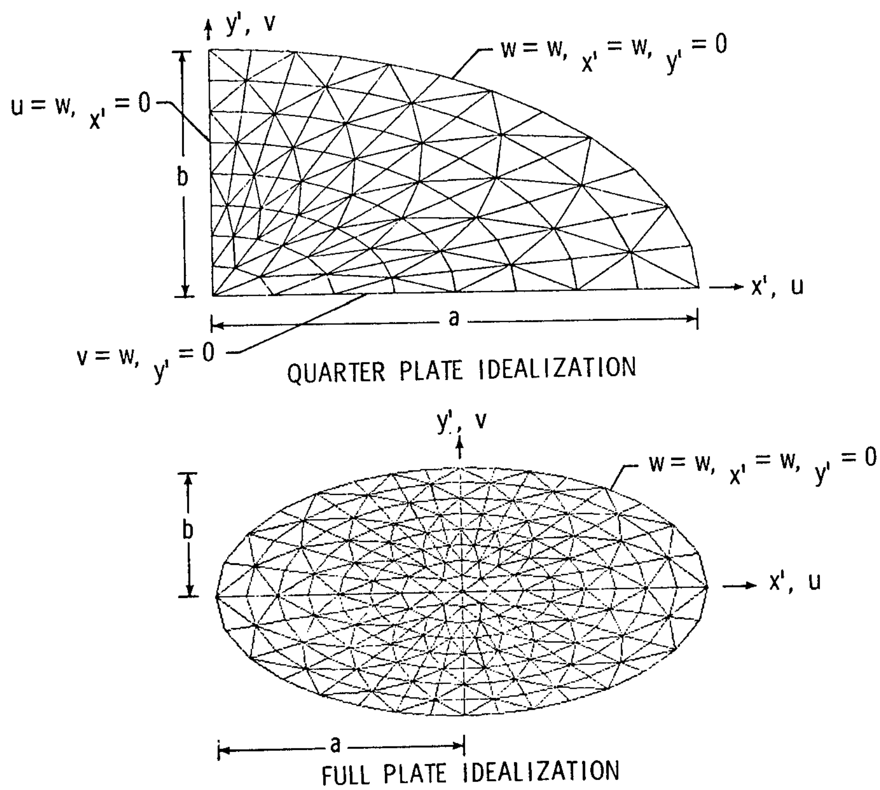


BOUNDARY FORCES IN TYPICAL SUBLAMINATES

LAMINATE STRAIN = -10^{-5}, h = .51 mm



FINITE ELEMENT IDEALIZATION OF A SUBLAMINATE



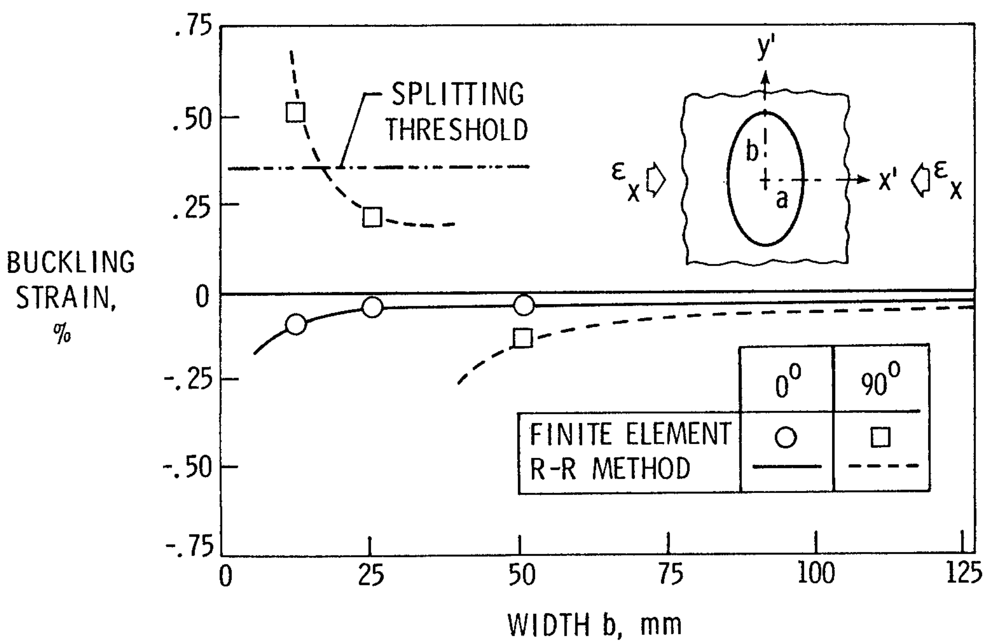
RAYLEIGH - RITZ METHOD

- DISPLACEMENT FUNCTION $w = \left[1-(x'/a)^2 - (y'/b)^2\right] \left\{C_0 + C_1 (x')^2 + C_2 (y')^2\right\}$
- TOTAL POTENTIAL ENERGY, $\Pi = U + V$
- TREFFTZ CRITERION $\delta^2 \Pi / \delta C_i^2 = 0$

$$\left| \left[K\right] - \varepsilon_{xc} \left[K_g\right] \right| = 0$$

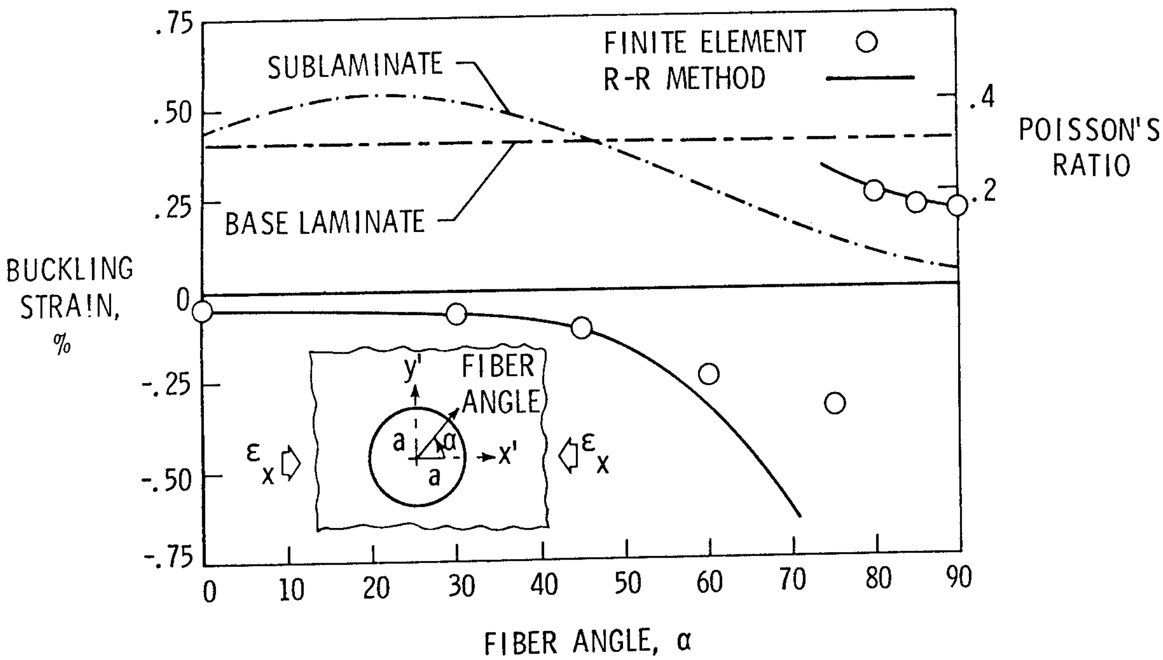
BUCKLING OF SPECIALLY ORTHOTROPIC SUBLAMINATES

$a = 25.4$, $h = .51$ mm, GRAPHITE/EPOXY



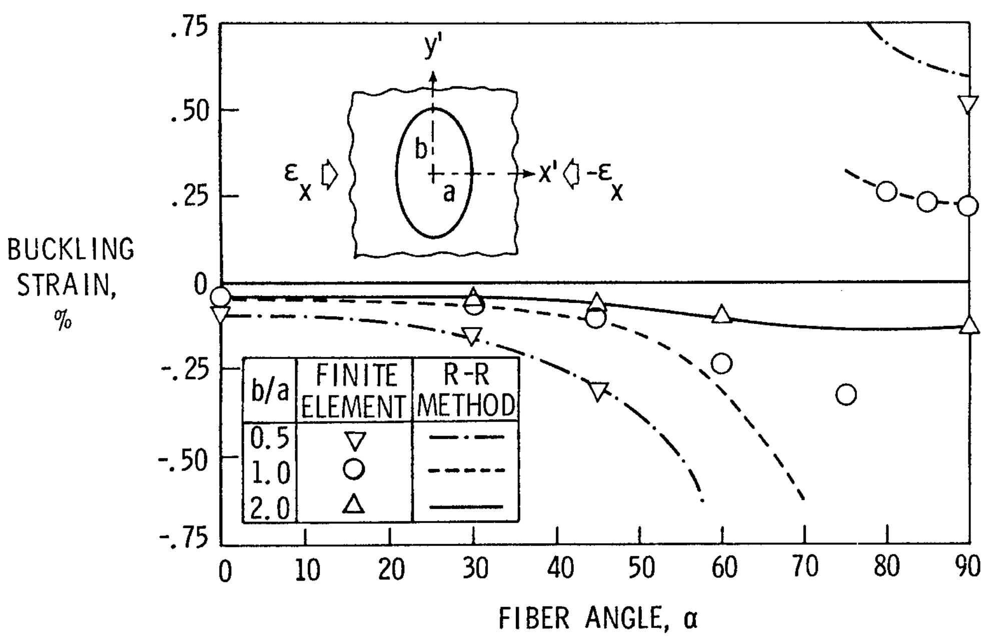
EFFECT OF FIBER ANGLE ON BUCKLING STRAIN

a = 25.4 mm, h = .51 mm



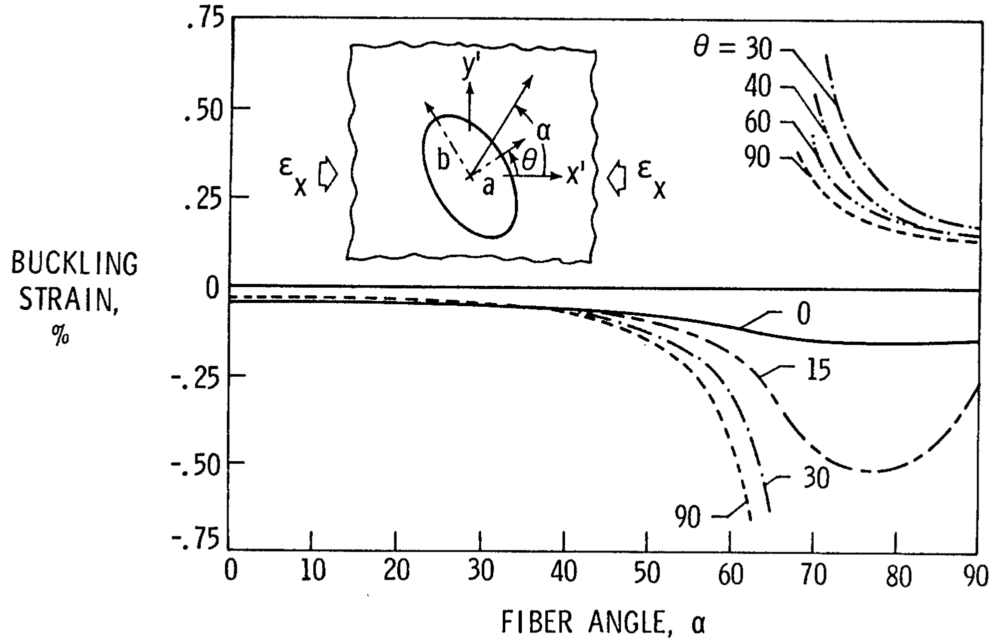
EFFECT OF FIBER ANGLE AND SUBLAMINATE SHAPE

a = 25.4 mm, h = .51 mm



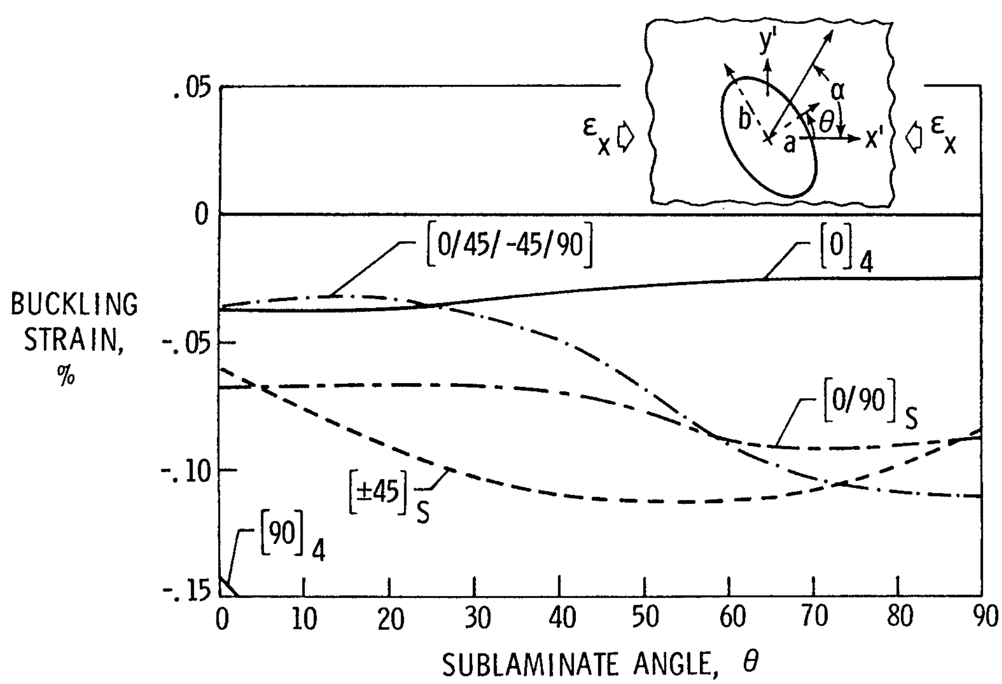
EFFECT OF FIBER ANGLE AND SUBLAMINATE ORIENTATION

a = 25.4, b = 50.8, h = .51 mm



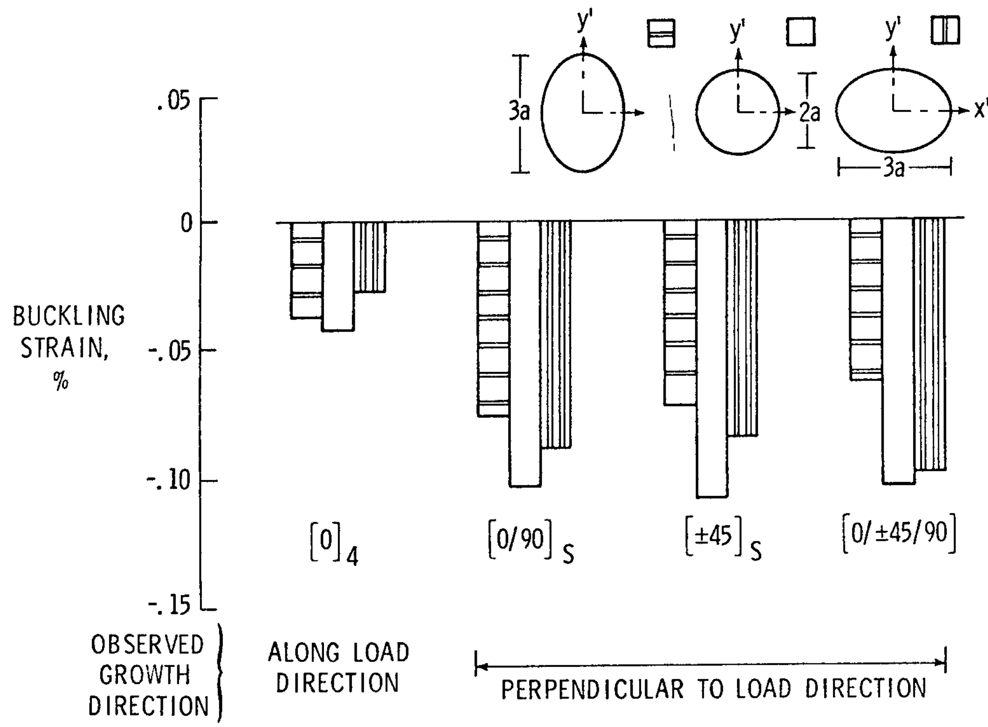
EFFECT OF SUBLAMINATE ORIENTATION

a = 25.4, b = 50.8, h = .51 mm



VARIATION OF BUCKLING STRAIN WITH DELAMINATION SHAPE

a = 25.4 mm, h = .51 mm, GRAPHITE/EPOXY



SUMMARY

- SUBLAMINATES ARE SUBJECTED TO GENERAL BIAXIAL STRESS STATE
- DEVELOPED ANALYSES TO PREDICT THE BUCKLING OF A SURFACE DELAMINATION IN A QUASI-ISOTROPIC LAMINATE
- UNIDIRECTIONAL COMPOSITE SUBLAMINATES CAN BUCKLE UNDER REMOTE COMPRESSION AS WELL AS TENSION STRAIN
- BUCKLING STRAINS OF UNIDIRECTIONAL COMPOSITE SUBLAMINATES (b > a) INCREASES WITH FIBER ANGLE
- BUCKLING STRAINS OF MULTIDIRECTIONAL FIBER SUBLAMINATES ARE BOUNDED BY 0^0 AND 90^0 FIBER SUBLAMINATES
- UNIDIRECTIONAL FIBER SUBLAMINATES GROWS IN THE DIRECTION OF LOAD; WHEREAS $(0/90)_S$, AND $(\pm 45)_S$, AND (0/±45/90) SUBLAMINATES GROW PERPENDICULAR TO LOAD DIRECTION

APPLICATION OF OPTIMIZATION TECHNIQUES TO COMPOSITE LAMINATES

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OBJECTIVES

DEVELOP EASY-TO-USE LAMINATE SIZING PROGRAMS THAT RUN ON MICROCOMPUTERS.
DESIGN VARIABLES:

- PLY RATIOS
- PLY ORIENTATIONS
- ORTHOTROPIC AXIS

DEMONSTRATE EFFECTIVENESS OF PROGRAMS

COMPARE EFFICIENCY OF DIFFERENT DESIGN VARIABLES

NON-LINEAR OPTIMIZATION METHODS

PLY RATIOS

MODIFIED METHOD OF FEASIBLE DIRECTIONS TAKES ADVANTAGE OF ANALYTIC TOTAL THICKNESS SCALING

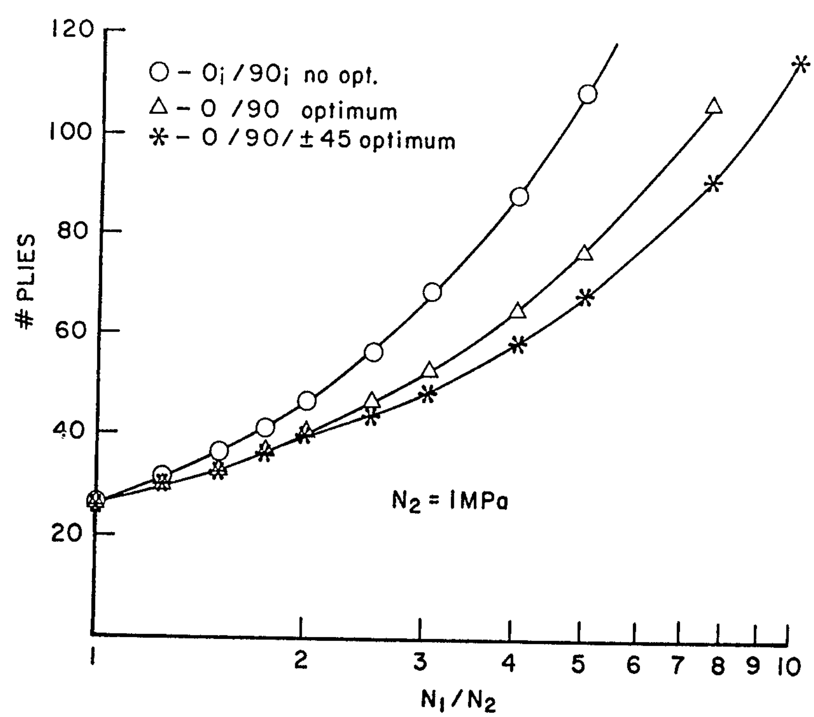
PLY ORIENTATIONS

INDIRECT METHOD - SEARCH DIRECTION BASED ON MINIMIZING A RELATED UNCONSTRAINED FUNCTION. VARIANCE OF ALL CONSTRAINTS FOUND TO BE MOST EFFECTIVE FUNCTION

ORTHOTROPIC ORIENTATION

ONE-DIMENSIONAL SEARCH FOR BEST RIGID BODY ROTATION. OPTIMIZE PLY RATIOS AT EACH STEP. APPROXIMATE FAILURE CRITERIA USED FOR SPEED.

$$\varepsilon_1^2 + \varepsilon_2^2 + \tfrac{1}{2}\varepsilon_6^2 \le b^2$$



TOTAL THICKNESS REQUIRED TO SUPPORT A SINGLE LOAD FOR VARIOUS π/n LAMINATES

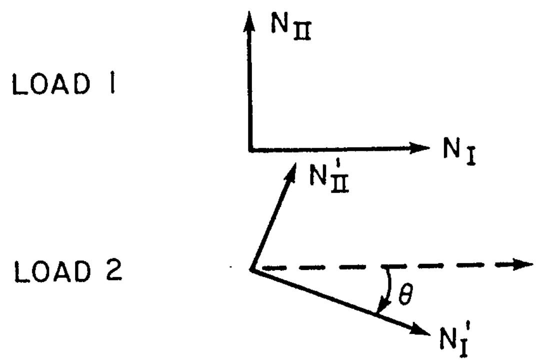
N1= 3 MN/M
N2= 1 MN/M
N6= 0 MN/M

$\Delta\theta$	# PLY GROUPS	TOTAL # OF PLIES
60	3	52
45	4	49
30	6	51
18	10	50
10	18	51

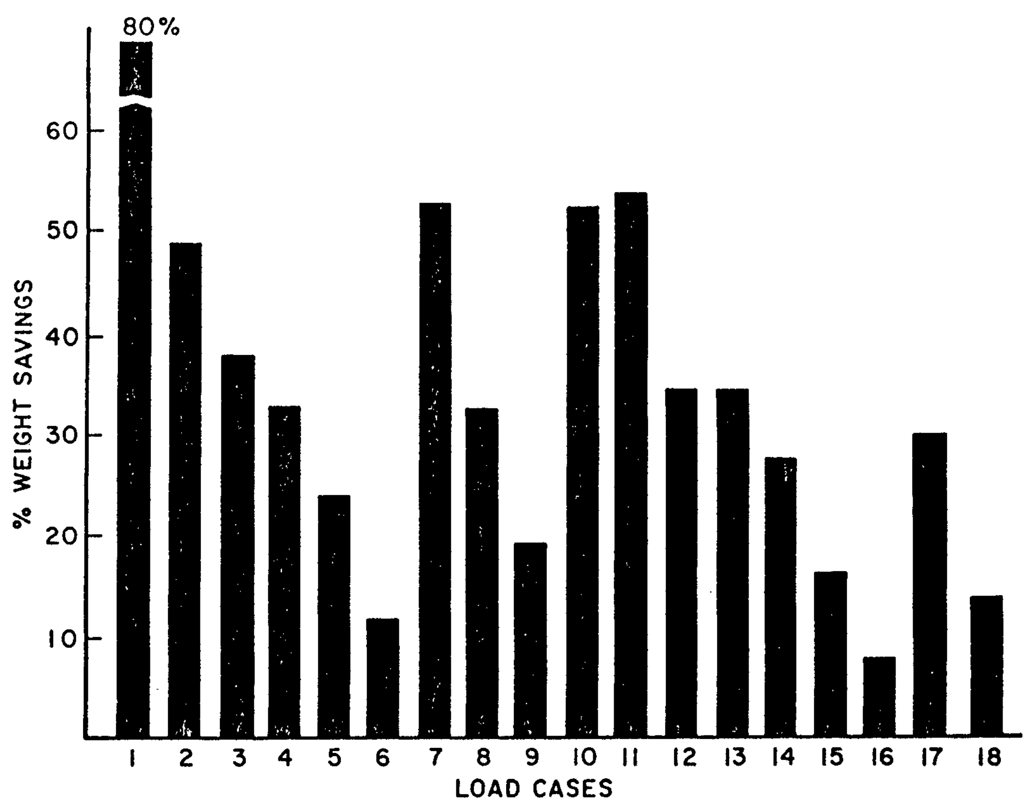
LOADS

N_1= 3 MN/M $\quad$ N_1'= 1.5 MN/M
N_2= 1 MN/M $\quad$ N_2'= 1.5 MN/M
N_6= 0 MN/M $\quad$ N_6'= -.5 MN/M

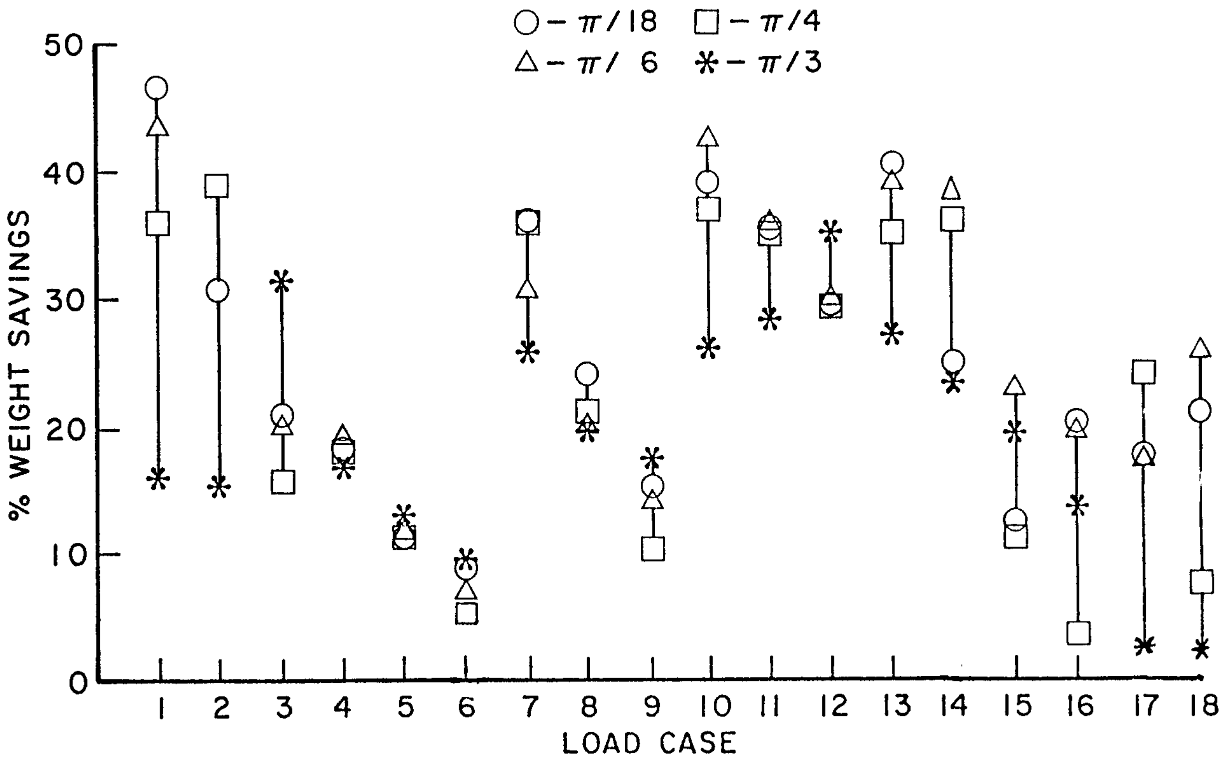
DESIGN VARIABLE	ANGLE	# PLIES	STRENGTH RATIO
PLY RATIOS	0	28.8	1.33/1.04
TOTAL # PLIES=	90	9.9	1.00/1.35
57.9	45	7.1	1.21/1.00
	-45	12.1	1.06/1.45
PLY ANGLES	-5	14.8	1.32/1.10
TOTAL # PLIES=	95	14.8	1.01/1.43
59.2	25	14.8	1.35/1.00
	-25	14.8	1.20/1.28
BALANCED LAMINATE	0	25.3	
TOTAL # PLIES= 59.3	90	7.2	
RIGID BODY ROTATION=	45	13.4	1.00/1.00
-6°	-45	13.4	

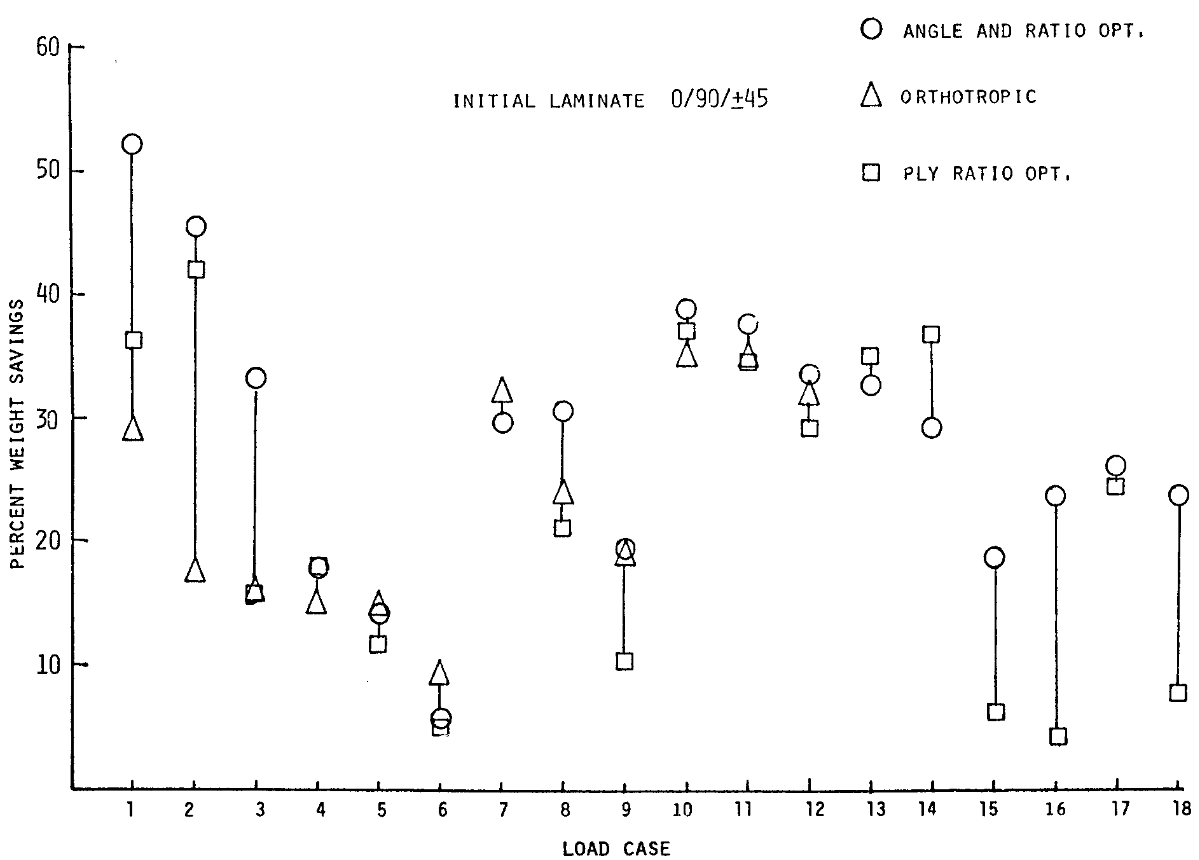


$N_I : N_{II}$	$N'_I : N'_{II}$	θ = 20°	40°	60°
1:0	1:0	1	2	3
2:1	2:1	4	5	6
4:1	4:1	7	8	9
2:1	4:1	10	11	12
−2:−1	−2:−1	13	14	15
−2:−1	−2:1	16	17	18
		LOAD CASE No.		



Weight Savings of Optimized (0/90/±45) over (0_i/90_i/±45_i) no opt.


○ − π/18
□ − π/4
△ − π/6
✱ − π/3
% WEIGHT SAVINGS
0
10
20
30
40
50
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18
LOAD CASE


INITIAL LAMINATE 0/90/±45
○ ANGLE AND RATIO OPT.
△ ORTHOTROPIC
□ PLY RATIO OPT.
PERCENT WEIGHT SAVINGS
10
20
30
40
50
60
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18
LOAD CASE

APPROXIMATE AND QUADRATIC CRITERIA FOR OPTIMIZATION

LOADS

N1= 4 MN/M	N1'= 2.76 MN/M
N2= 1 MN/M	N2'= 2.24 MN/M
N6= 0 MN/M	N6'= 1.48 MN/M

PLY GROUP		# PLIES NEEDED TSAI-WU	APPROXIMATE
0		35.2	35.2
90		7.5	7.4
45		9.9	10.8
-45		33.8	33.7
	TOTAL	86.5	87.1

OPTIMALITY CRITERION

derived from

$$\varepsilon_1^2 + \varepsilon_2^2 + \tfrac{1}{2}\varepsilon_6^2 \le b^2$$

$$\vec{\varepsilon}^{\,T}|T||A^{-1}||Q^{(\theta_i)}|\vec{\varepsilon} = \lambda$$

λ is a constant for each ply group

$$T = \begin{vmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & \frac{1}{2} \end{vmatrix}$$

CONCLUSIONS

- A PACKAGE OF EFFICIENT AND USER-FRIENDLY LAMINATE SIZING PROGRAMS IS AVAILABLE THROUGH AFWAL/MLBM
- WEIGHT SAVINGS OVER QUASI-ISOTROPIC LAMINATES ARE USUALLY LARGE (20 - 30%) EVEN FOR MULTIPLE BIAXIAL LOADING CONDITIONS
- [0/90/±45] LAMINATES SEEM TO BE A GOOD STARTING POINT FOR ANY COMBINATION OF LOADS
- THE DESIGNER HAS A CHOICE OF DESIGN VARIABLES WHICH WILL EACH GIVE EQUALLY EFFICIENT LAMINATES
- AN APPROXIMATE FAILURE CRITERIA HAS BEEN SHOWN TO GIVE GOOD RESULTS WHILE DRASTICALLY REDUCING COMPUTATION TIME.

COMPOSITE MECHANICS/RELATED ACTIVITIES AT LEWIS RESEARCH CENTER

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NASA LEWIS RESEARCH CENTER
CLEVELAND, OHIO 44135

NINTH ANNUAL MECHANICS OF COMPOSITES REVIEW
DAYTON, OHIO, OCTOBER 24-26, 1983

OBJECTIVE

SUMMARY OF LEWIS RESEARCH ACTIVITIES AND PROGRESS IN:

- o COMPOSITE MECHANICS
- o COMPUTER PROGRAMS FOR COMPOSITES
- o HIGH TEMPERATURE COMPOSITES
- o COMPOSITE ENGINE COMPONENTS

CONCLUSIONS

- o CURRENT LEWIS RESEARCH ACTIVITIES ON COMPOSITE MECHANICS/RELATED AREAS INCLUDE:
  - o COMPOSITE MECHANICS, COMPUTER PROGRAMS FOR COMPOSITES, HIGH TEMPERATURE COMPOSITES AND COMPOSITE ENGINE STRUCTURAL COMPONENTS
  - o RECENT PROGRESS INCLUDES:
    - o SIMPLIFIED MICROMECHANICS EQUATIONS WITH AND WITHOUT INTERPHASE
    - o FINITE ELEMENT SUBSTRUCTURING FOR COMPOSITE MECHANICS
    - o APPLICATION OF THE LEWIS LIFE/DURABILITY THEORY
    - o FAILURE MODES LONGITUDINAL COMPRESSION IMPACT AND DYNAMIC DELAMINATION
    - o COMPLETION OF INHYD
    - o DEVELOPMENT OF: ICAN, CODSTRAN, N. L. COBSTRAN, STAEBL
    - o INITIATION OF RESEARCH IN HIGH TEMPERATURE COMPOSITE AND HIGH-STRAIN RATE EFFECTS ON STRESS CONCENTRATION AND ENVIRONMENTAL BEHAVIOR
    - o STRUCTURAL TAILORING OF COMPOSITE FAN BLADES
    - o THERMOVISCOPLASTIC STRUCTURAL ANALYSIS OF TURBINE BLADES MADE FROM TUNGSTEN-FIBER REINFORCED SUPERALLOYS

LEWIS RESEARCH ACTIVITIES IN COMPOSITE MECHANICS/RELATED AREAS

- o COMPOSITE MECHANICS
  - o MICROMECHANICS - SIMPLIFIED EQUATIONS
  - o F. E. SUBSTRUCTURING AND SPECIALTY FINITE ELEMENTS
  - o LIFE/DURABILITY
  - o FAILURE MODES
- o COMPUTER PROGRAMS FOR COMPOSITES
  - o INHYD
  - o ICAN
  - o CODSTRAN
  - o N. L. COBSTRAN
  - o STAEBL
- o HIGH TEMPERATURE COMPOSITES
  - o TEST METHODS AND CHARACTERIZATION (UP TO 2000°F)
  - o COMPOSITE BURNER LINERS
  - o TUNGSTEN-FIBER REINFORCED SUPERALLOYS (FRS)
- o COMPOSITE ENGINE STRUCTURAL COMPONENTS
  - o COMPOSITE FRAMES
  - o FAN BLADES - SUPERHYBRID, WITH COMPOSITE INLAYS
  - o SWEPT TURBOPROPS AND PROPS FOR GENERAL AVIATION AIRCRAFT
  - o FRS TURBINE BLADES

COMPOSITE MECHANICS

- o SIMPLIFIED MICROMECHANICS EQUATIONS/F. E. VALIDATION
- o SIMPLIFIED MICROMECHANICS EQUATIONS WITH INTERPHASE
- o F. E. SUBSTRUCTURING IN COMPOSITE MECHANICS AND LAMINATE ANALYSIS
- o LONGITUDINAL COMPRESSION BEHAVIOR-FAILURE MODES
- o LIFE/DURABILITY IN HYGROTHERMOMECHANICAL ENVIRONMENTS
- o DEVELOPMENT OF HYGROTHERMOMECHANOCHRONIC THEORY
- o DYNAMIC INTERPLY DELAMINATION
- o HIGH-STRAIN-RATE EFFECTS ON STRESS CONCENTRATION AND ENVIRONMENTAL BEHAVIOR
- o DEVELOPMENT OF SPECIALTY FINITE ELEMENTS FOR NONLINEAR, TRANSIENT SHELL ANALYSIS

COMPUTER PROGRAMS FOR COMPOSITES

- o INHYD — INTRAPLY HYBRID COMPOSITE DESIGN
- o ICAN — INTEGRATED COMPOSITES ANALYSIS
- o CODSTRAN — COMPOSITE STRUCTURAL DURABILITY STRUCTURAL ANALYSIS
- o N. L. COBSTRAN — NONLINEAR COMPOSITE BLADE STRUCTURAL ANALYSIS
- o STAEBL — STRUCTURAL TAILORING OF ENGINE BLADES

HIGH TEMPERATURE COMPOSITES

- TEST METHODS AND CHARACTERIZATION
- COMPOSITE BURNER LINER
- TUNGSTEN-FIBER REINFORCED SUPERALLOYS

COMPOSITE ENGINE STRUCTURAL COMPONENTS

- COMPOSITE FRAMES
- FAN BLADES - SUPERHYBRID, COMPOSITE INLAYS
- SWEPT TURBOPROPS
- COMPOSITE BLADES FOR GENERAL AVIATION AIRCRAFT ENGINES

COMPOSITE MICROMECHANICS

MECHANICAL PROPERTIES

LONGITUDINAL MODULUS: $E_{\ell 11} = k_f E_{f11} + k_m E_m$

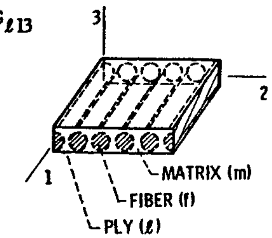
TRANSVERSE MODULUS: $E_{\ell 22} = \frac{E_m}{1 - \sqrt{k_f}(1 - E_m/E_{f22})} = E_{\ell 33}$

SHEAR MODULUS: $G_{\ell 12} = \frac{G_m}{1 - \sqrt{k_f}(1 - G_m/G_{f12})} = G_{\ell 13}$

SHEAR MODULUS: $G_{\ell 23} = \frac{G_m}{1 - k_f(1 - G_m/G_{f23})}$

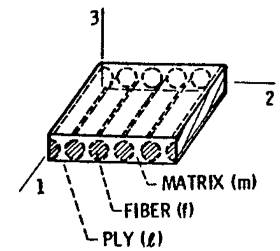
POISSON'S RATIO: $\nu_{\ell 12} = k_f \nu_{f12} + k_m \nu_m = \nu_{\ell 13}$

POISSON'S RATIO: $\nu_{\ell 23} = \frac{E_{\ell 22}}{2G_{\ell 23}} - 1$



COMPOSITE MICROMECHANICS

THERMAL PROPERTIES



HEAT CAPACITY: $C_\ell = \frac{1}{\rho_\ell}(k_f \rho_f C_f + k_m \rho_m C_m)$

LONGITUDINAL CONDUCTIVITY: $K_{\ell 11} = k_f K_{f11} + k_m K_m$

TRANSVERSE CONDUCTIVITY: $K_{\ell 22} = (1 - \sqrt{k_f}) K_m + \frac{K_m \sqrt{k_f}}{1 - \sqrt{k_f}(1 - K_m/K_{f22})} = K_{\ell 33}$

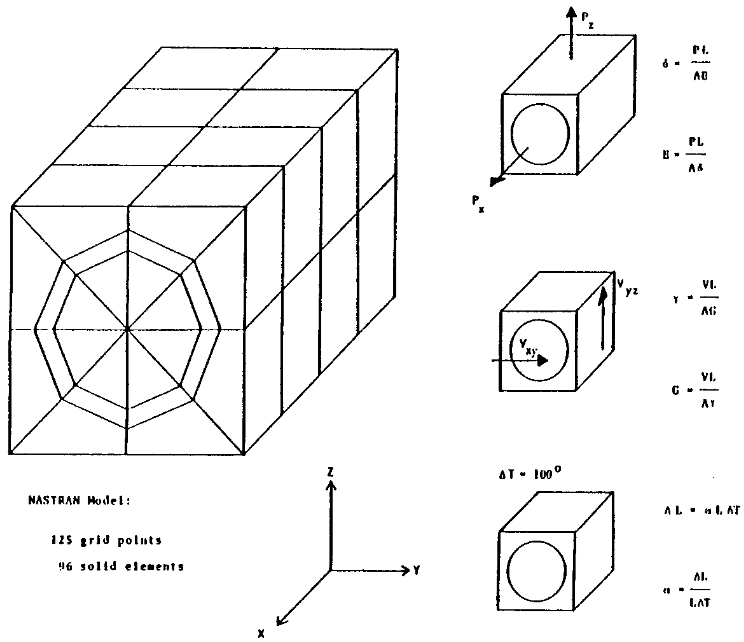
FOR VOIDS: $K_m = (1 - \sqrt{k_v}) K_m + \frac{K_m \sqrt{k_v}}{1 - \sqrt{k_v}(1 - K_m/K_v)}$

LONGITUDINAL THERMAL EXPANSION COEFFICIENT: $\alpha_{\ell 11} = \frac{k_f \alpha_{f11} E_{f11} + k_m \alpha_m E_m}{E_{\ell 11}}$

TRANSVERSE THERMAL EXPANSION COEFFICIENT: $\alpha_{\ell 22} = \alpha_{f22}\sqrt{k_f} + (1 - \sqrt{k_f})(1 + k_f \nu_m E_{f11}/E_{\ell 11})\alpha_m$
$= \alpha_{\ell 33}$

MICROMECHANICS/NASTRAN VALIDATION

FINITE ELEMENT TEST MODEL



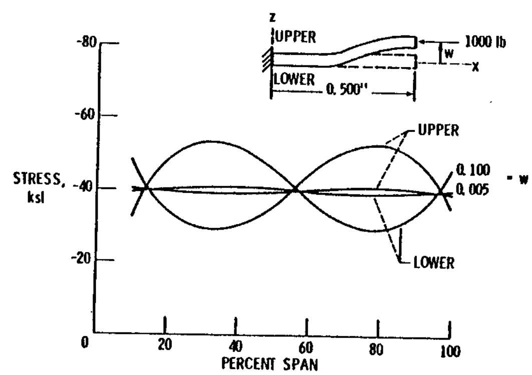
MICROMECHANICS/NASTRAN VALIDATION

RESULTS

PROPERTY	MICROMECHANICS PREDICTION	NASTRAN RESULTS
$E_{\ell 11}$	37.2	37.2
$E_{\ell 22}$	35.7	35.3
$G_{\ell 12}$	13.8	14.3
$G_{\ell 23}$	13.8	14.0
$\nu_{\ell 12}$	0.296	0.297
$\nu_{\ell 23}$	0.296	0.278
$\alpha_{\ell 11}$	3.96	4.00
$\alpha_{\ell 22}$	4.98	4.34

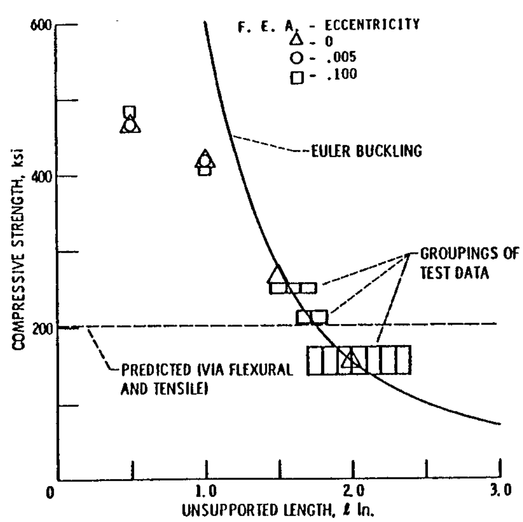
ECCENTRICITY AFFECTS STRESS VARIATION ALONG COLUMN

(T300/E, UDC, 0.10 In. THICK)



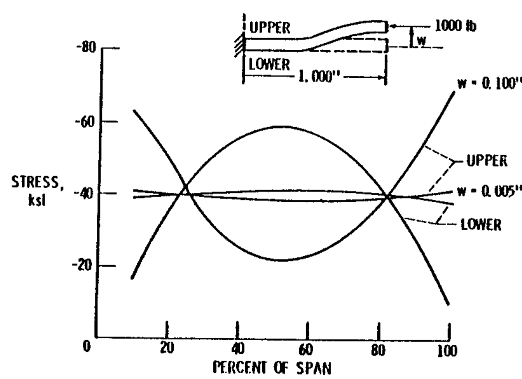
PROGRESSIVE END-TAB DEBONDING SEVERELY AFFECTS EULER BUCKLING LOAD

(T300/E, 0.100 IN THICK)



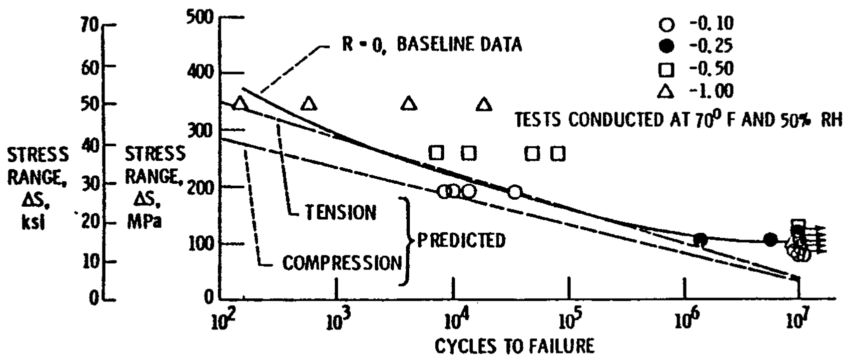
ECCENTRICITY AFFECTS STRESS VARIATION ALONG COLUMN

(T300/E, UDC, 0.10 In. THICK)

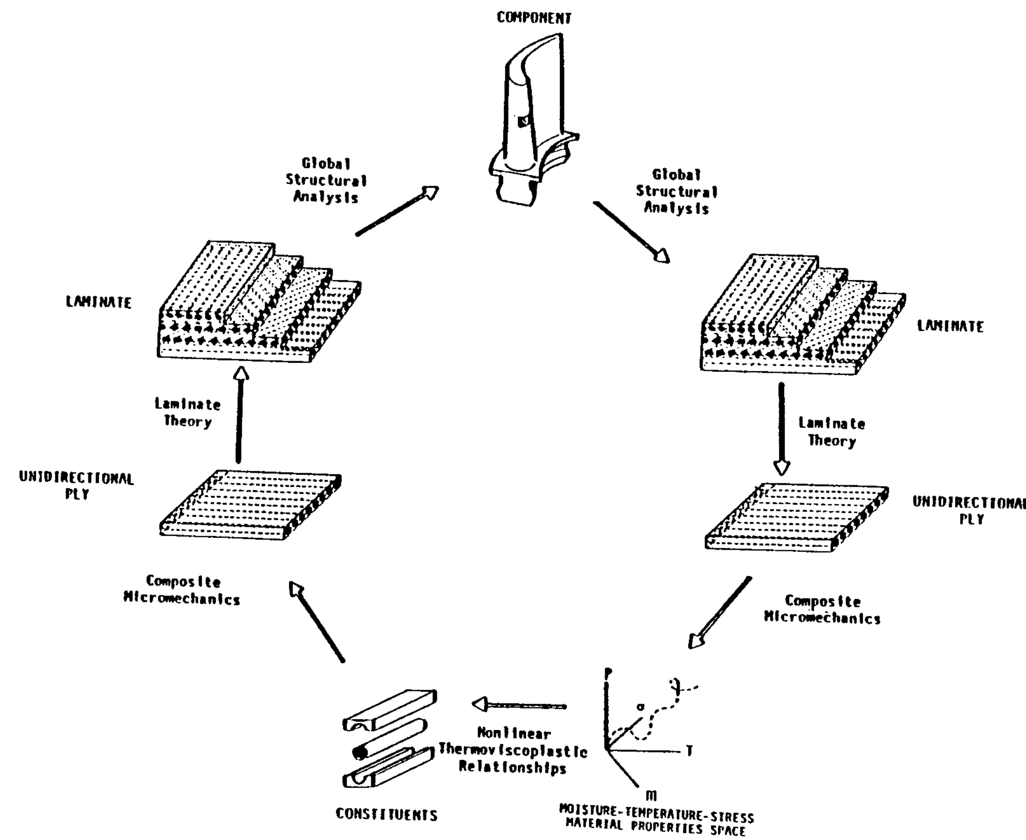


COMPARISON OF EFFECT OF MEAN STRESS ON FATIGUE ENDURANCE

WINDING PATTERN 1 TFT/EPOXY COMPOSITES

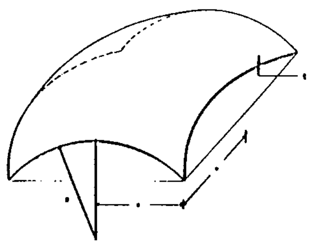


UPWARD INTEGRATED/"TOP-DOWN STRUCTURED" MECHANISTIC THEORY



2-D AND 3-D SPECIALTY FINITE ELEMENTS FOR NONLINEAR AND TRANSIENT STRUCTURAL ANALYSIS OF MULTILAYERED (THIN/THICK) FIBER COMPOSITE SHELL STRUCTURES

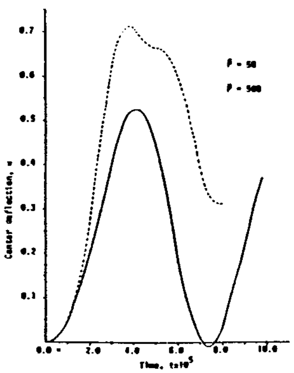
SPHERICAL CAP SCHEMATIC



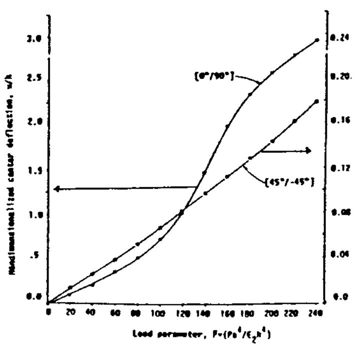
UNIFORM PRESSURE

R/a = 10

a/h = 100

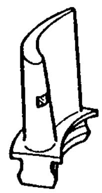


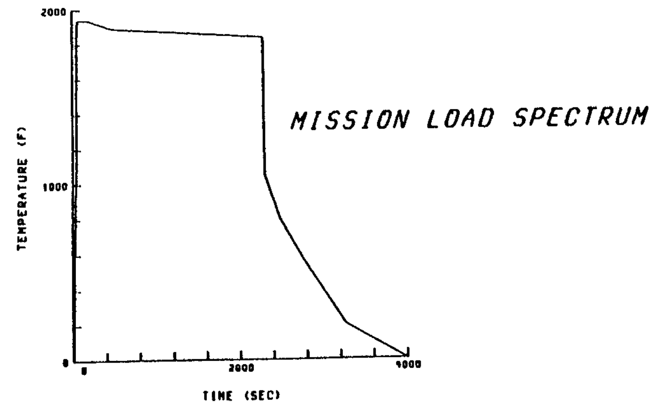
TRANSIENT RESPONSE OF A TWO-PLY FIBER COMPOSITE SHELL

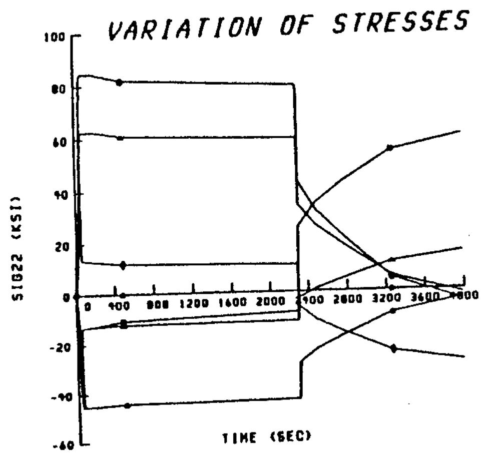


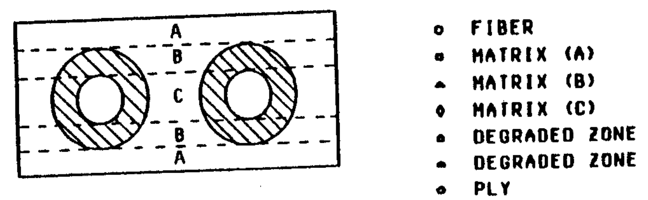
NONLINEAR RESPONSE OF TWO-PLY FIBER COMPOSITE SHELLS

THERMOVISCOPLASTIC NONLINEAR STRUCTURAL ANALYSIS OF FRS TURBINE BLADES
(PRELIMINARY RESULTS)

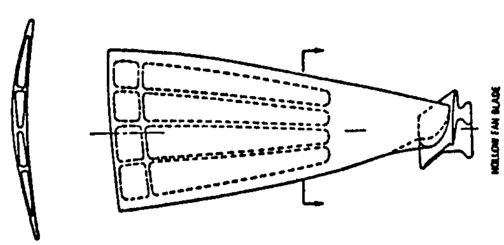




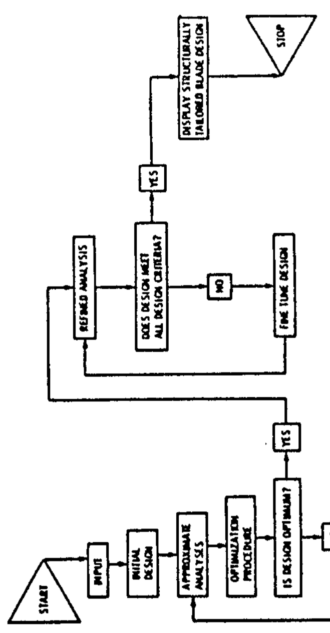




STRUCTURAL TAILORING OF ENGINE BLADES (STAEBL)



HOLLOW FAN BLADE



STAEBL FLOW CHART

FORMALIZED PROCEDURE FOR STRUCTURAL TAILORING (OPTIMUM DESIGN) OF ENGINE BLADES TO MEET AEROTHERMOMECHANICAL DESIGN REQUIREMENTS USING ADVANCED MATERIALS AND DESIGN CONCEPTS

STRUCTURALLY TAILORED SHROUDLESS BLADES
WITH COMPLEX INTERNAL STRUCTURE

PARAMETER	PERCENT CHANGE FROM REFERENCE DESIGN: HOLLOW BLADE WITH COMPOSITE INLAYS	PERCENT CHANGE FROM REFERENCE DESIGN: SUPERHYBRID COMPOSITE
. BLADE CHORD	-18	-9
. BLADE WEIGHT	-52	-37
. DIRECT OPERATIONAL COST PLUS INTEREST (DOC +I)		
. ENGINE WEIGHT	-0.33	-0.23
. ENGINE COST	-0.15	-0.18
. MAINTENANCE COST	+0.03	+0.05
TOTAL	-0.45	-0.36

DESIGN REQUIREMENTS SATISFIED:
- RESONANCE MARGINS (FIRST, SECOND, THIRD, AND FOURTH MODES)
- FLUTTER-LOG DECREMENT (FIRST, SECOND, AND THIRD MODES)
- BIRD INGESTION (LOCAL AND ROOT STRESSES)
- STEADY STATE STRESS (THROUGHOUT THE BLADE)

STATISTICAL EVALUATION OF FAILURE DATA FOR COMPOSITE MATERIALS

DONALD M. NEAL AND LUCIANO SPIRIDIGLIOZZI

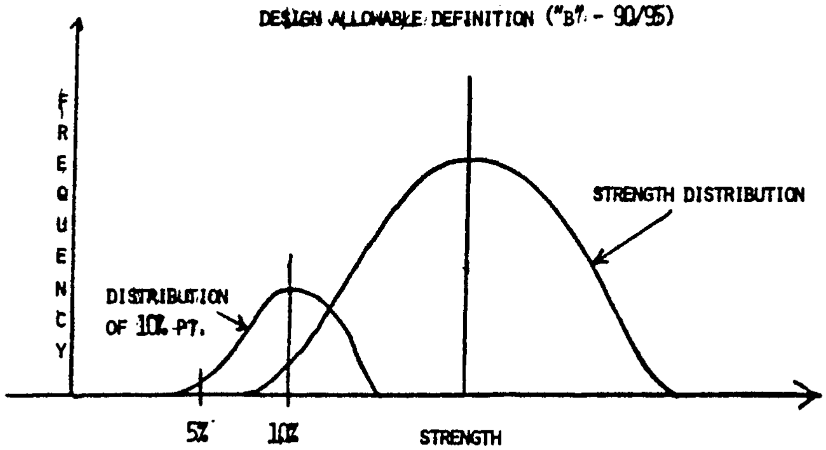
ARMY MATERIALS AND MECHANICS RESEARCH CENTER
WATERTOWN, MA

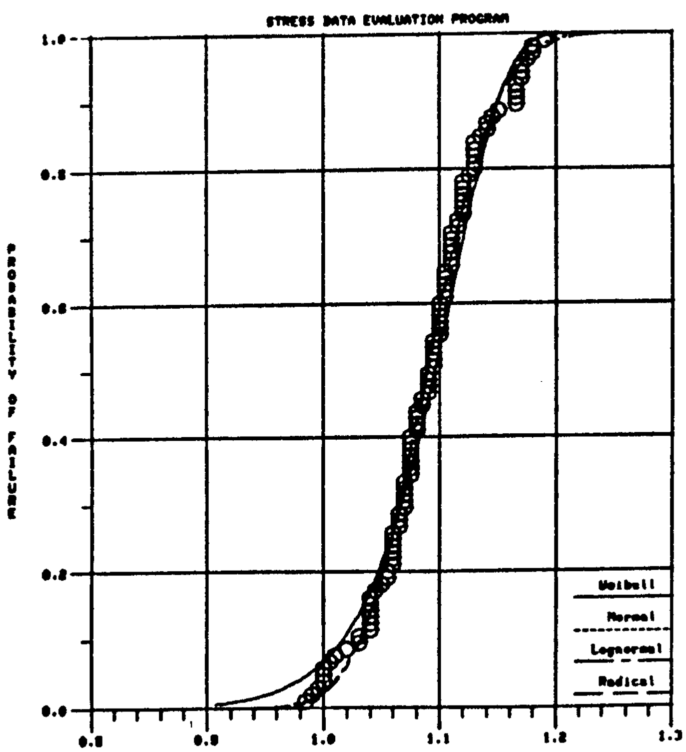
OBJECTIVES

- TO DEVELOP EFFICIENT METHODS FOR OBTAINING DESIGN ALLOWABLES
- TO DEMONSTRATE THE VALUE OF EXPLORATORY DATA METHODOLOGY MATERIAL IN STATISTICAL MODELLING OF COMPOSITE MATERIAL STRENGTH DATA
- TO EVALUATE THE ACCEPTABILITY OF CONVENTIONAL AND NEW TECHNIQUES IN DETERMINING THE ALLOWABLE VALUES

CONCLUSIONS

- EXPLORATORY DATA PROCEDURE SHOULD BE APPLIED PRIOR TO ACCEPTANCE OF STATISTICAL MODEL USED IN THE ALLOWABLE COMPUTATION
- QUANTILE BOX PLOT PROVIDES AN EXCELLENT SUMMARY OF TEST DATA RESULTS IN ADDITION TO LOCATING OUTLIERS AND MULTI-MODALITY IN THE SAMPLE
- A NEW METHOD FOR ESTIMATING TAIL PROBABILITIES AND EXTREME VALUE DISTRIBUTIONS DEVELOPED BY L. BREIMAN, PROVED TO BE THE MOST DESIRABLE ALLOWABLE ESTIMATES PROCEDURE
- THE AUTHORS RECOMMEND NOT USING THE WEIBULL DISTRIBUTION FUNCTION FOR OBTAINING THE ALLOWABLES, IF OUTLIERS (HIGHER ORDERED VALUES) OR MULTI-MODALITY, EXIST IN DATA SET
- AN EXTREME VALUE DISTRIBUTION (BREIMAN METHOD) IF POOLED SAMPLES DO NOT REPRESENT GENERAL DATA POPULATION IS SUGGESTED
- THE INFORMATIVE QUANTILE FUNCTION WILL PROVIDE THE NECESSARY GUIDANCE IN SELECTING THE STATISTICAL MODELS
- IN MULTI-MODALITY CASE, CENSORED DATA ANALYSIS, BOOT-STRAP METHOD OR BREIMAN'S METHOD IS SUGGESTED FOR THE ALLOWABLE DETERMINATION
- AT PRESENT, THE AUTHORS RECOMMEND POOLING ALL SAMPLES MADE AVAILABLE EVEN THOUGH SIGNIFICANT DIFFERENCE TEST INDICATED OTHERWISE
- NON-PARAMETRIC PROCEDURES ARE ALWAYS THE MOST DESIRABLE IN OBTAINING THE ALLOWABLES FOR THE GIVEN SAMPLE IF PROPERLY APPLIED
- A SUFFICIENTLY LARGE NUMBER OF SOURCES IN OBTAINING TEST DATA IS MORE IMPORTANT IN DETERMINING ALLOWABLES THAN SIZE OF INDIVIDUAL SAMPLE





RMS ERROR

	Normal	Lognormal	Weibull	Radical
R1	.0208	.0221	.0422	.0299
R2	.0206	.0219	.0417	.0306
R3	.0205	.0218	.0410	.0298

		90% CONFIDENCE INTERVAL	
MEAN (KSI)=	1.090	1.082	1.098
S DEV (KSI)=	.046	.042	.052

WEIBULL PARAMETERS

		90% CONFIDENCE INTERVAL	
SLOPE M=	24.990	22.068	28.441
CHAR VALUE=	1.112	1.105	1.120
99% ORIGIN=	.927	.900	(95% CL)
ORIGIN=	.000		

RADICAL PARAMETERS

A=	.920	EXP B (N)=	12.000
B=	.274	EXP(R)=	2.000
C=	-.125	SIGF=	1.194
SIGI=	.795		

NON-PARAMETRI SOLN: A AND B ALLOWABLES

OPTIMAL:A(99,95,N=300)B(90,95,N= 30)
GIVEN DATA:
A(90,65,N=104)B(90,88,N=104)

WEIBULL:

DESIGN A= .940
DESIGN B= 1.001

NORMAL:

DESIGN A= .956
DESIGN B= 1.020

LOGNORMAL:

DESIGN A= .972
DESIGN B= 1.021

HLC0 + HBC0 + HAC0

Quantile Box Plot (Parzen)

Quantile function defined as

$$Q(u) = F^{-1}(u),\ 0 \le u \le 1$$

That is, if x random variable with distribution function given by F(x), then root of F(x) = u, 0 ≤ u ≤ 1 is p^{th} quantile of F(x).

From the ordered statistic $X_1 \le X_2 \le \ldots \le X_n$, Q is defined as piece wise linear function with interval (0, 1) divided into 2n subintervals

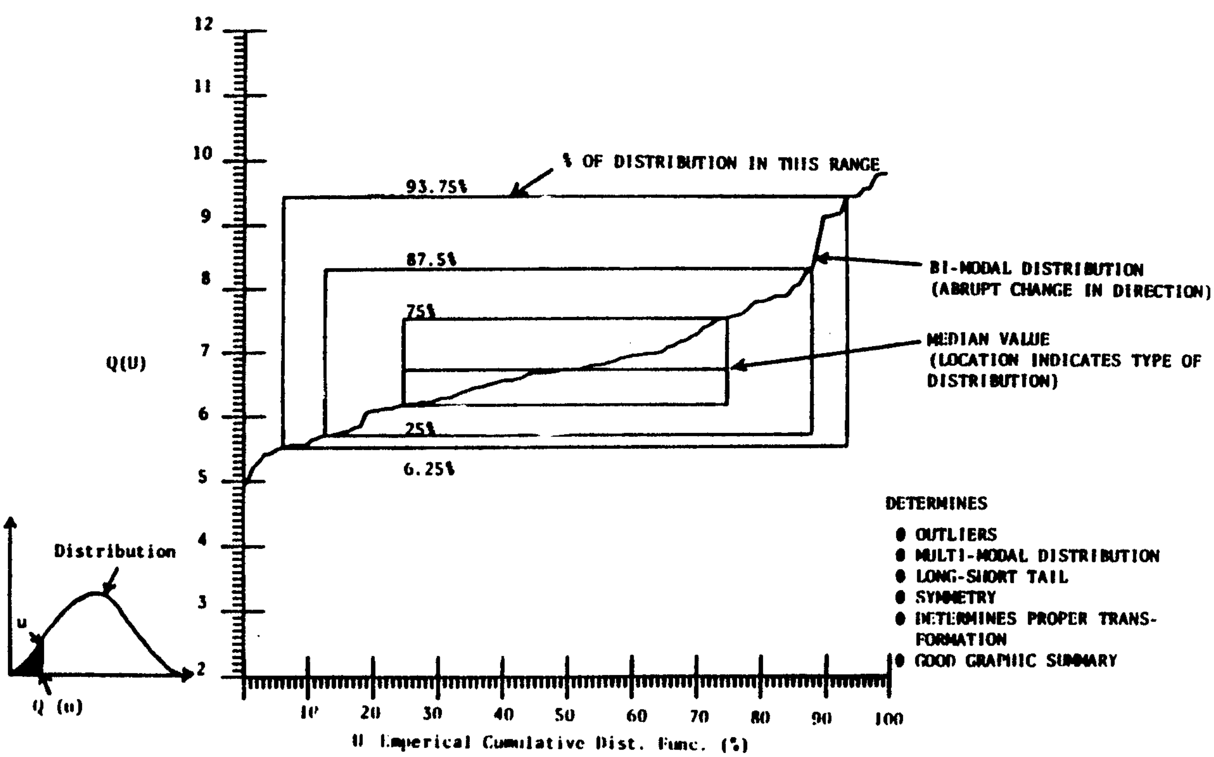
$$Q\left(\frac{2j-1}{2n}\right) = X_j \qquad j = 1, 2, \ldots, n$$

$$\text{for} \quad u \in \left[\frac{2j-1}{2n}, \frac{2j+1}{2n}\right]$$

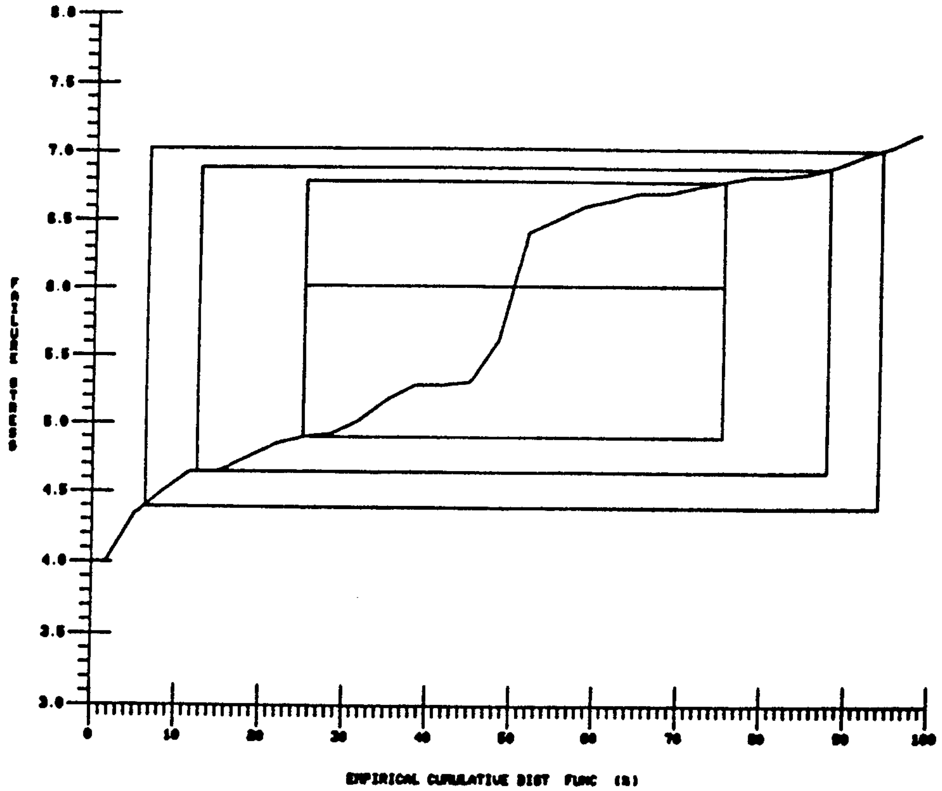
$$Q = n\left(u - \frac{2j-1}{2n}\right) X_{j+1} + n\left(\frac{2j+1}{2n} - u\right) X_j$$

Box Boundary

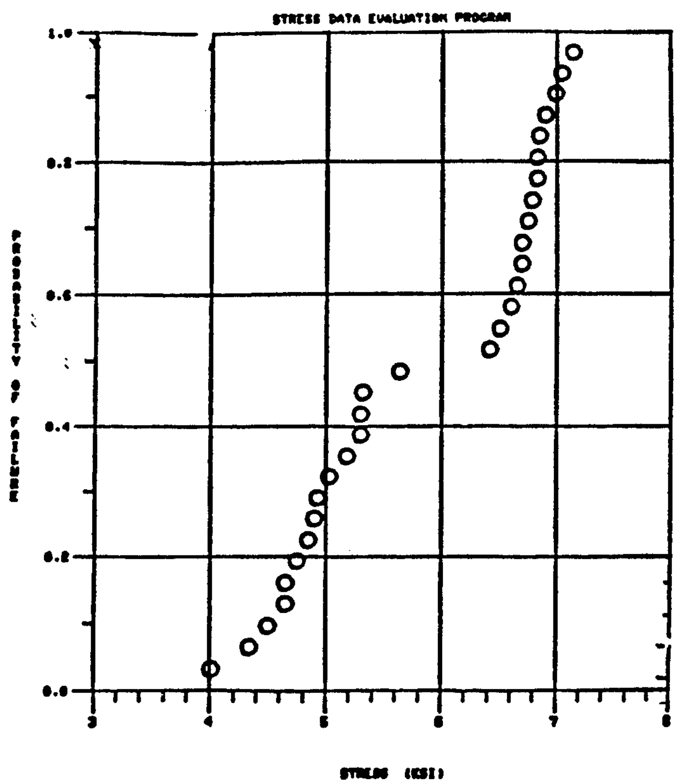
- Q (.25) to Q (.75)
- Q (.125) to Q (.875)
- Q (.0625) to Q (.9375)



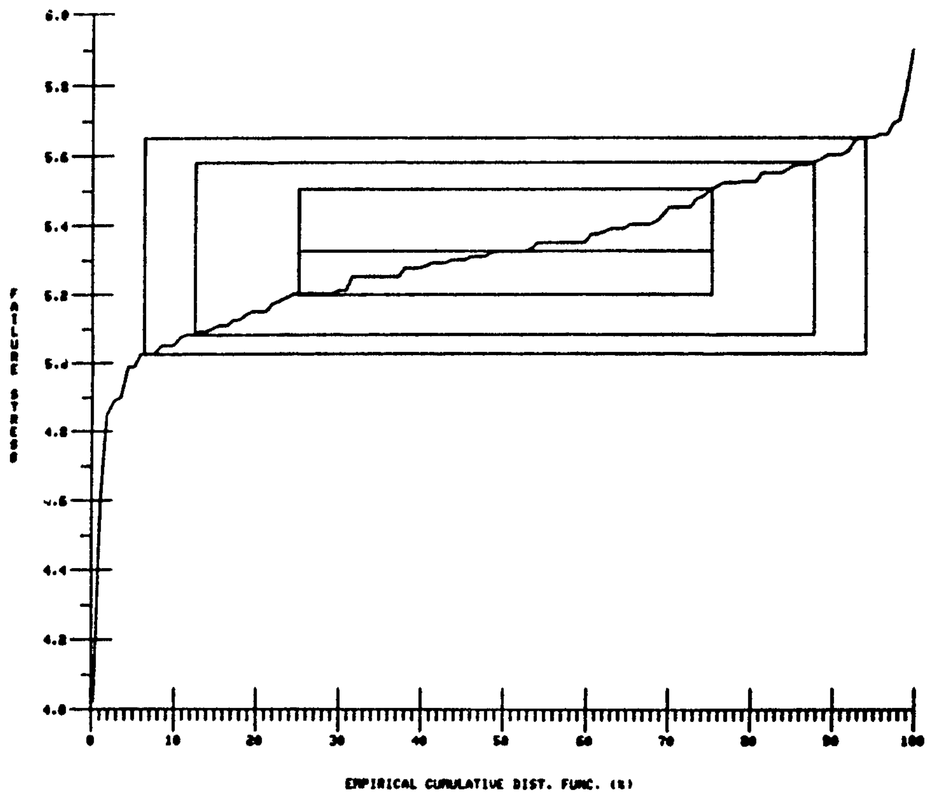
Quantile Box Plot

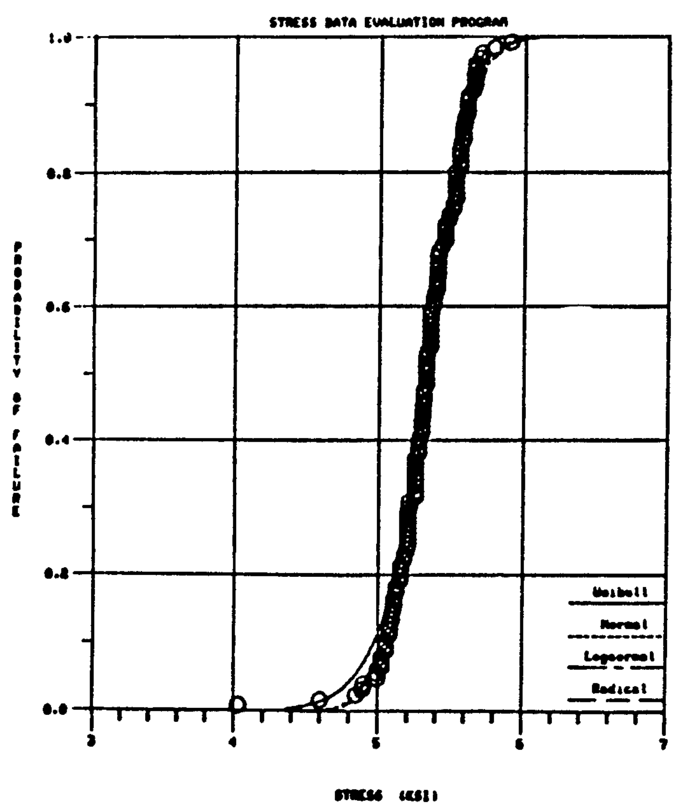


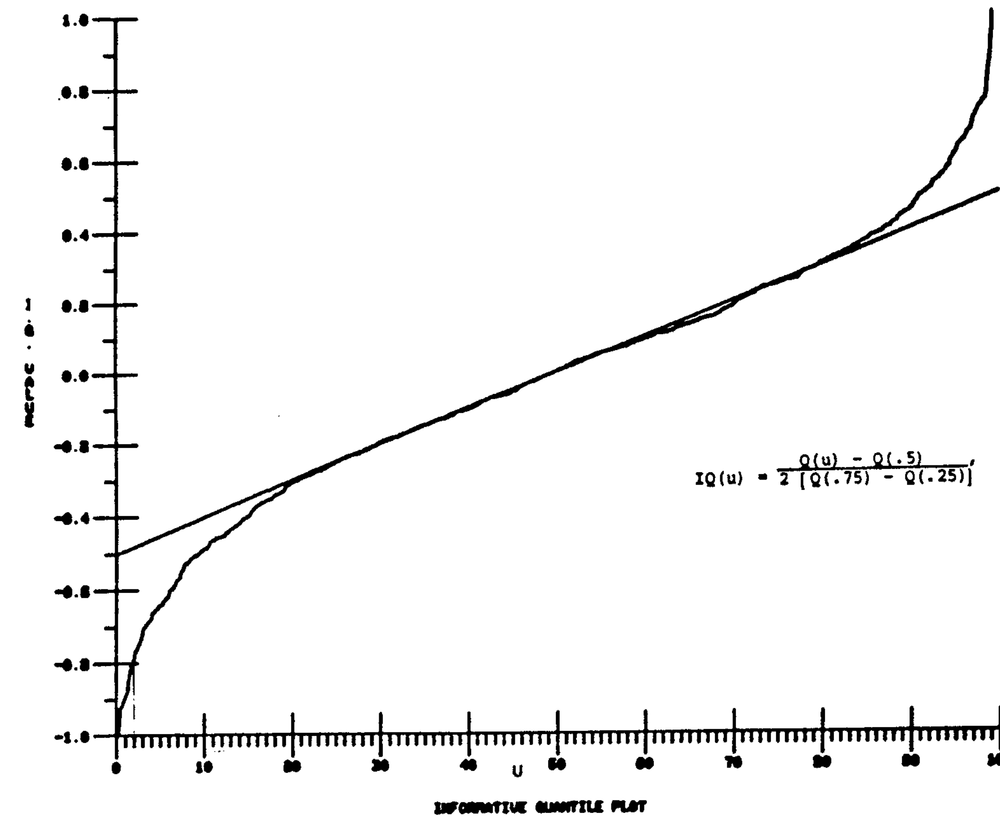
4.65
6.99
6.83
6.7
6.65
6.7
6.75
7.05
4.5
4.75
5.31
5.63
5.29
4.65
5.03
5.29
6.41
6.83
6.5
7.15
6.85
5.18
4.93
4.85
6.6
4.01
4.34
6.9
4.9
6.79



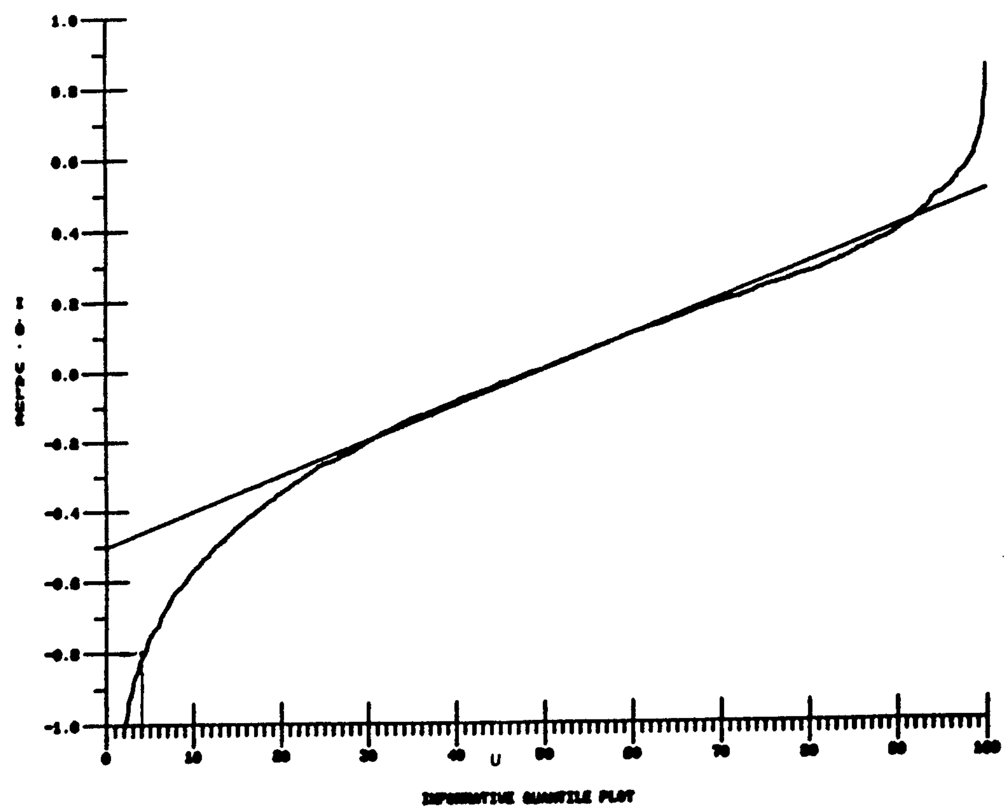
STATISTICALLY RANKED FAILURE RESULTS


FAILURE STRESS
EMPIRICAL CUMULATIVE DIST. FUNC. (%)

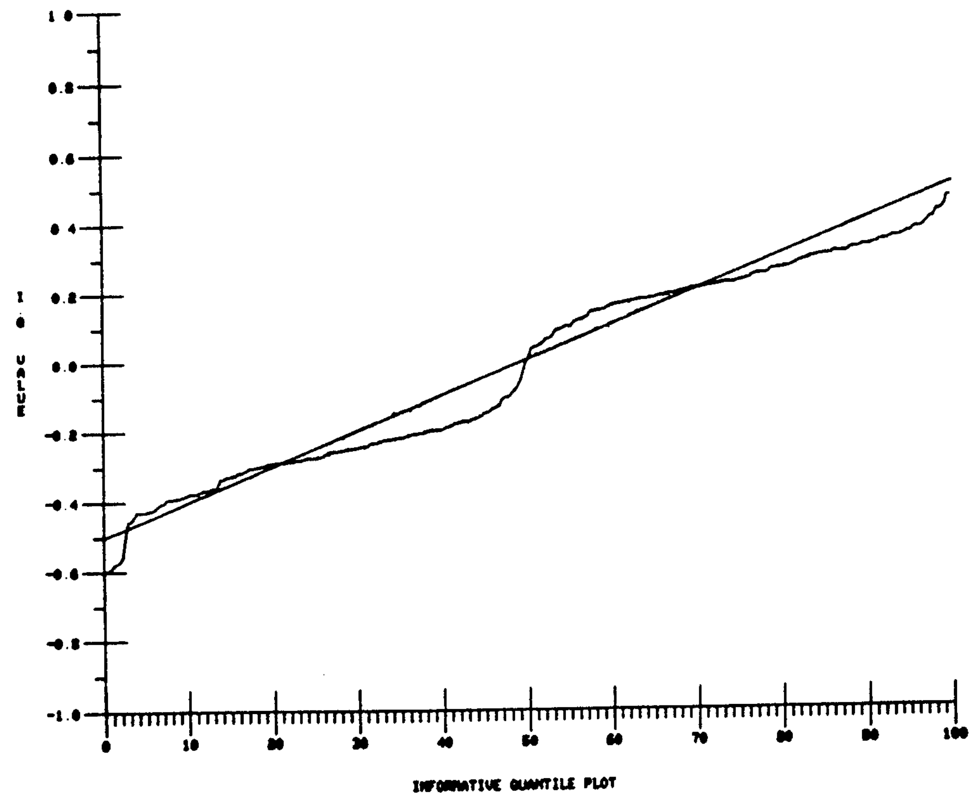

STRESS DATA EVALUATION PROGRAM
PROBABILITY OF FAILURE
STRESS (KSI)

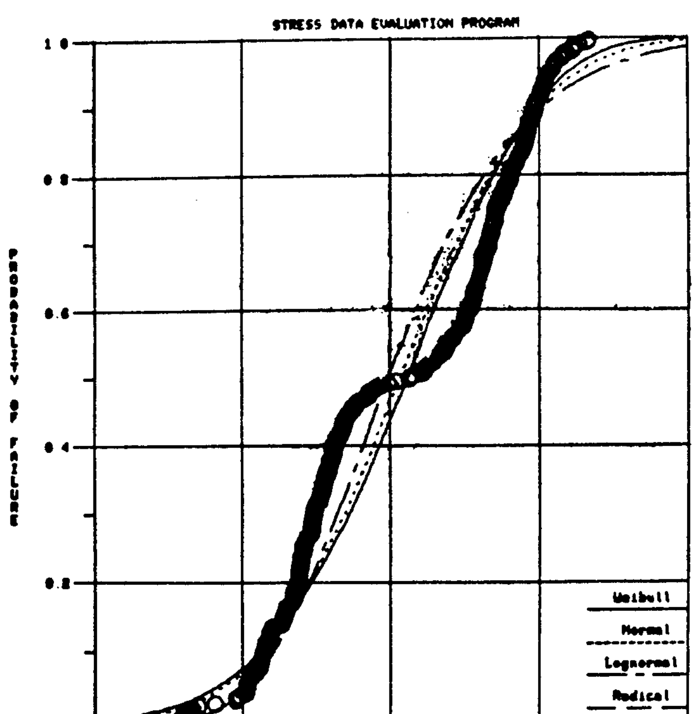


NORMAL FUNCTION



WEIBULL FUNCTION





RMS ERROR

	Normal	Lognormal	Weibull	Radical
R1	.0791	.0849	.0781	.0583
R2	.0792	.0854	.0783	.0581
R3	.0794	.0852	.0784	.0588

		95% CONFIDENCE INTERVAL	
MEAN (KSI)=	5.670	5.983	6.880
S DEV (KSI)=	1.468	1.367	1.584

WEIBULL PARAMETERS

		95% CONFIDENCE INTERVAL	
SLOPE M=	4.881	4.330	5.408
CHAR VALUE=	6.718	6.496	6.931
99% ORIGIN=	2.686	2.431	(95% CL)
ORIGIN=	.000		

RADICAL PARAMETERS

A=	5.663	EXP B (N)=	2.000
B=	3.530	EXP(R)=	2.000
C=	-2.006	SIGF=	9.083
SIG(I)=	8.896		

NON-PARAMETRIC SOLN: A AND B ALLOWABLES

OPTIMAL A(99,95,N=300)B(90,95,N= 30)

GIVEN DATA A(99,95,N=300)B(90,95,N=300)

WEIBULL:

DESIGN A= 2.431
DESIGN B= 4.046

NORMAL:

DESIGN A= 2.468
DESIGN B= 4.070

LOGNORMAL:

DESIGN A= 3.174
DESIGN B= 4.182

BREIMAN METHOD

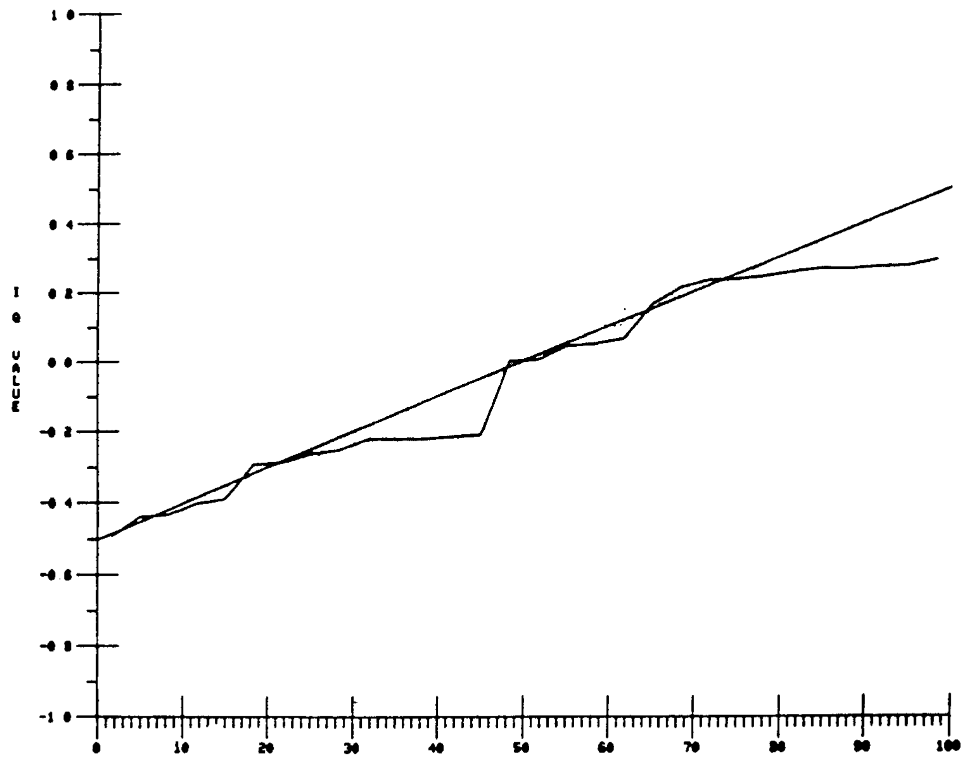
"A" Allowable = 3.12

"B" Allowable = 4.14

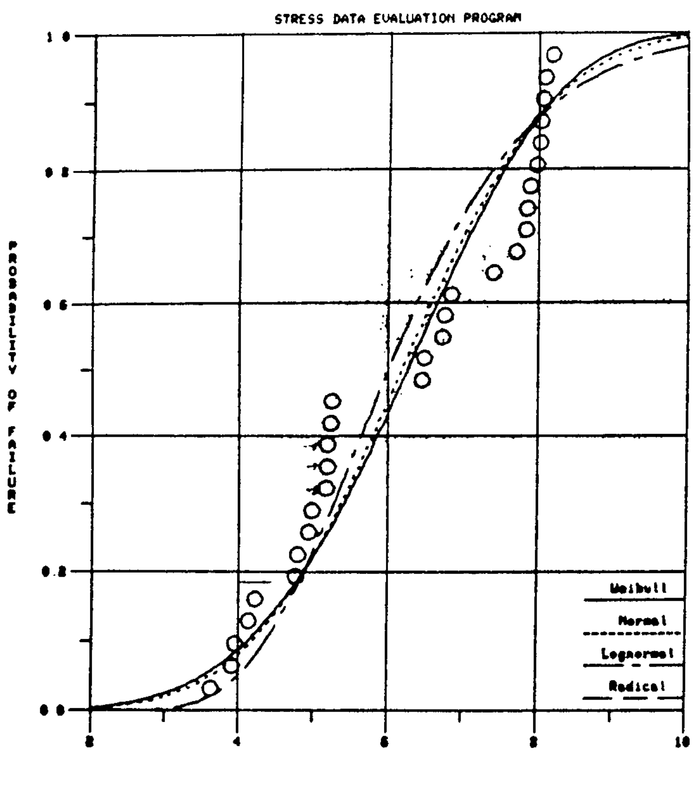
NON-PARAMETRIC

"A" = 3.14

"B" = 4.16



INFORMATIVE QUANTILE PLOT



RMS ERROR

	Normal	Lognormal	Weibull	Radical
R1	0829	0893	0806	0691
R2	0839	0906	0815	0673
R3	0856	0926	0831	0652

		95% CONFIDENCE INTERVAL	
MEAN (KSI)=	6 189	5 695	6 683
S DEV (KSI)=	1 565	1 314	2 038

WEIBULL PARAMETERS

		95% CONFIDENCE INTERVAL	
SLOPE M=	4 500	3 633	5 747
CHAR VALUE=	6 790	6 386	7 205
95% ORIGIN=	2 568	1 791	(95% CL)
ORIGIN=	000		

RADICAL PARAMETERS

A= 4 785 EXP B (N)= 2 000
B= 4 506 EXP(R)= 2 000
C= -2.382 SIGF= 0 231
SIGI= 2.403

NON-PARAMETRI SOLN: A AND B ALLOWABLES

OPTIMAL A(99.95.N=300)B(90.95.N= 30)
GIVEN DATA:
A(99.95.N= 30)B(90.95.N= 30)

WEIBULL:

DESIGN A= 1.791
DESIGN B= 3 457

NORMAL:

DESIGN A= 1 393
DESIGN B= 3 407

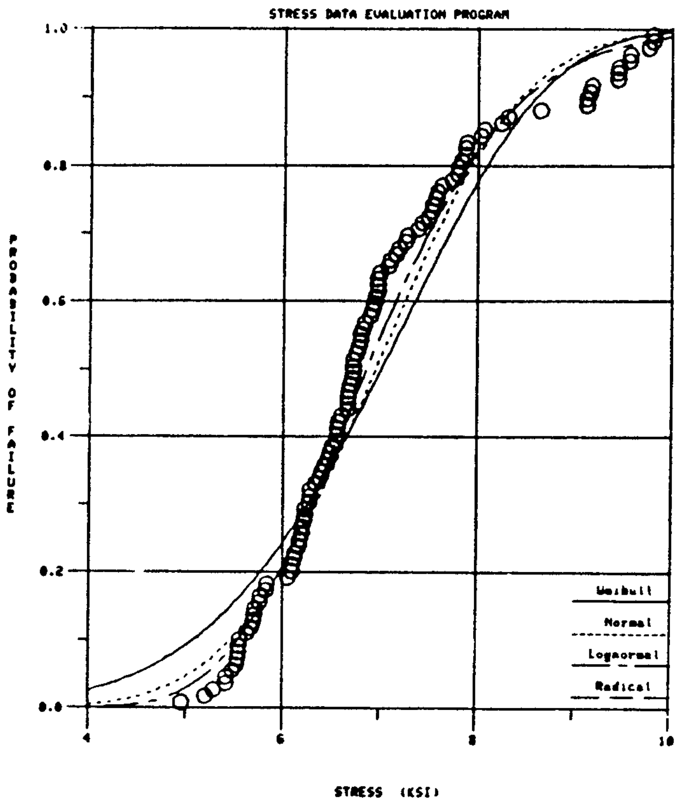
LOGNORMAL:

DESIGN A= 2 640
DESIGN B= 3.731

BREIMAN METHOD

"A" Allowable = 2.30

"B" Allowable = 3.80



RMS ERROR

	Normal	Lognormal	Weibull	Radical
R1	.0589	.0401	.0766	.0248
R2	.0587	.0398	.0760	.0251
R3	.0584	.0395	.0752	.0257

		90% CONFIDENCE INTERVAL	
MEAN (KSI)=	6.952	6.762	7.143
S DEV (KSI)=	1.162	1.046	1.310

WEIBULL PARAMETERS

		90% CONFIDENCE INTERVAL	
SLOPE M=	5.858	5.187	6.646
CHAR VALUE=	7.458	7.251	7.672
99% ORIGIN=	3.433	3.030	(95% CL)
ORIGIN=	.000		

RADICAL PARAMETERS

A=	16.306	EXP B (N)=	12.000
B=	3.624	EXP(R)=	12.000
C=	-13.746		
SIGI=	2.560	SIGF=	10.930

NON-PARAMETRI SOLN: A AND B ALLOWABLES
OPTIMAL:A(99,95,N=300)B(90,95,N= 30)
GIVEN DATA:
A(99,66,N=100)B(90,88,N=100)

WEIBULL:
DESIGN A= 3.030
DESIGN B= 4.758

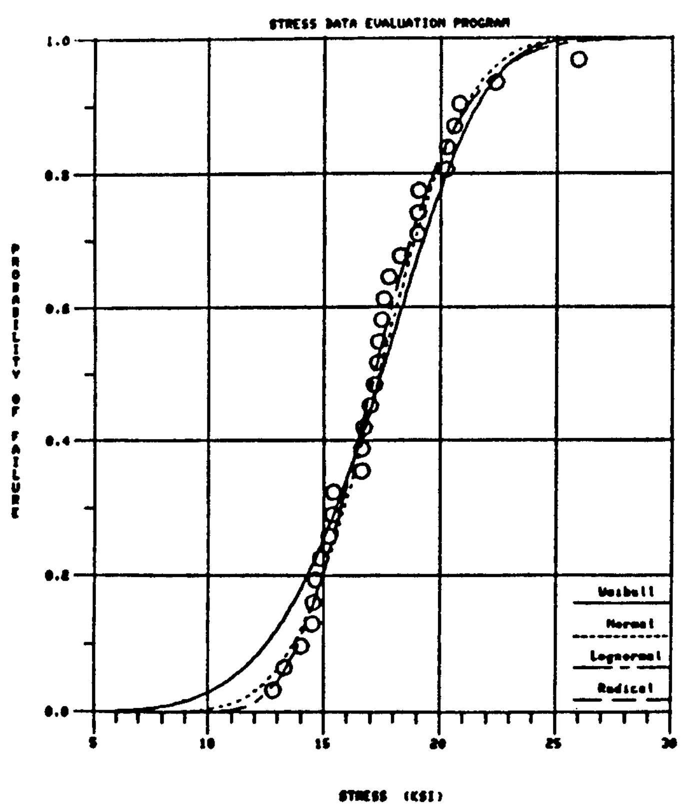
NORMAL:
DESIGN A= 3.852
DESIGN B= 5.190

LOGNORMAL:
DESIGN A= 4.477
DESIGN B= 5.383

BREIMAN METHOD

"A" Allowable = 4.19

"B" Allowable = 5.56



RMS ERROR

	Normal	Lognormal	Weibull	Radical
R1	.0434	.0329	.0640	.0335
R2	.0424	.0321	.0613	.0331
R3	.0416	.0318	.0675	.0329

		90% CONFIDENCE INTERVAL	
MEAN (KSI)=	17.381	16.473	18.289
S DEV (KSI)=	2.877	2.416	3.745

WEIBULL PARAMETERS

		90% CONFIDENCE INTERVAL	
SLOPE M=	5.663	4.445	7.231
CHAR VALUE=	18.616	17.596	19.700
99% ORIGIN=	8.570	6.454	(95% CL)
ORIGIN=	.000		

RADICAL PARAMETERS

A=	30.478	EXP B (N)=	12.000
B=	11.329	EXP(R)=	12.000
C=	-35.189		
SIGI=	4.289	SIGF=	50.807

NON-PARAMETRI SOLN: A AND B ALLOWABLES
OPTIMAL:A(99,95,N=300)B(90,95,N= 30)
GIVEN DATA:
A(99,26,N= 30)B(90,96,N= 30)

WEIBULL:
DESIGN A= 6.454
DESIGN B= 10.887

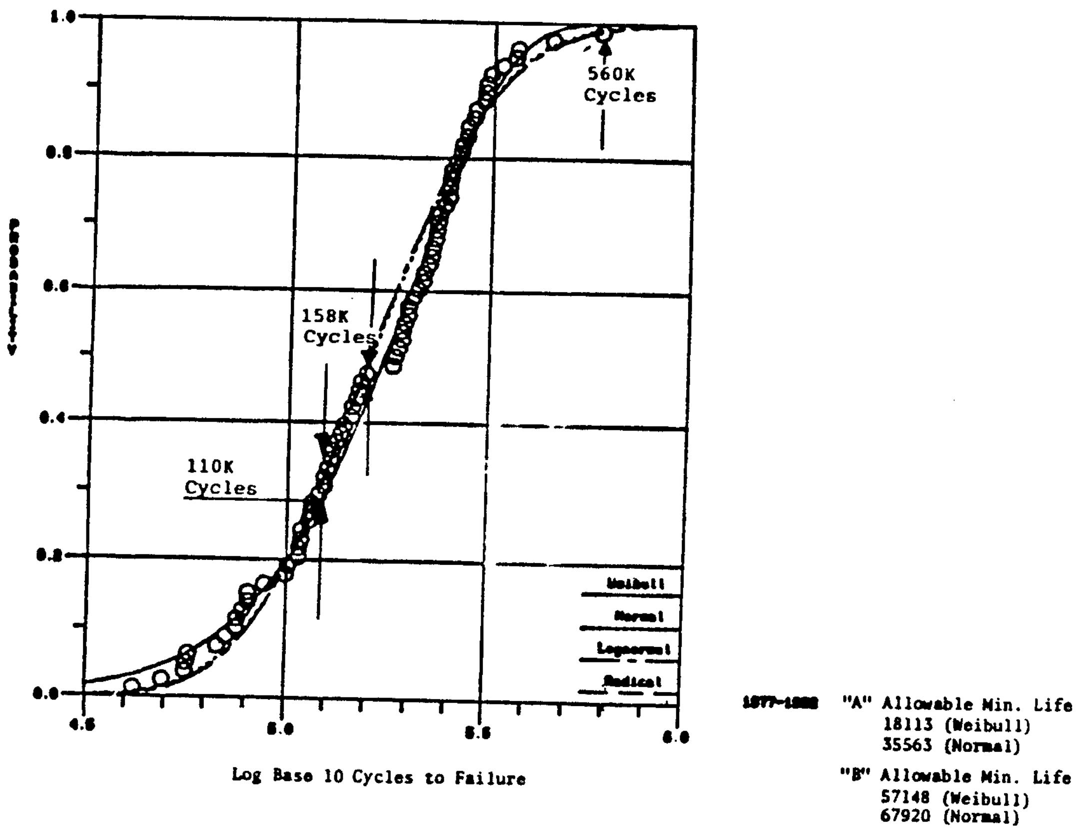
NORMAL:
DESIGN A= 8.565
DESIGN B= 12.268

LOGNORMAL:
DESIGN A= 10.520
DESIGN B= 12.921

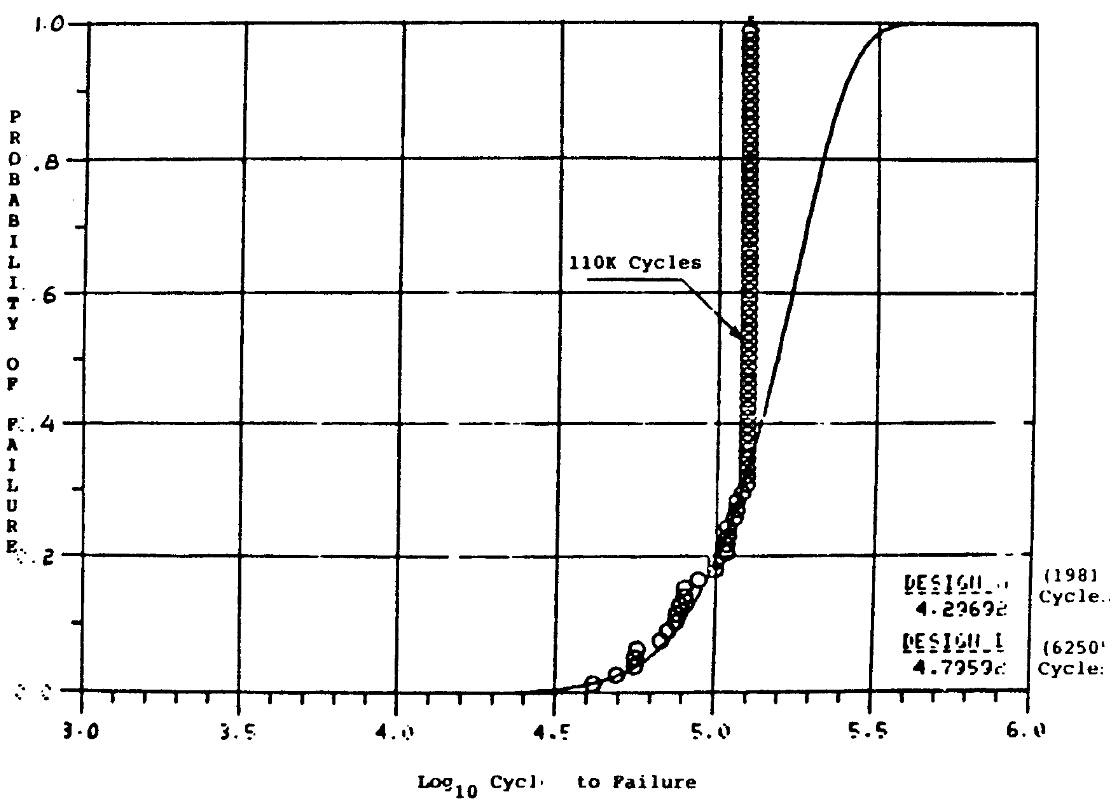
BREIMAN METHOD

"A" = 9.71

"B" = 13.40



Weibul-Censored Data Analysis of Scranton Torsional Fatigue Test Results



GLOBAL-LOCAL MODEL

FOR

LAMINATE ANALYSIS

N. J. PAGANO

&

S. R. SONI

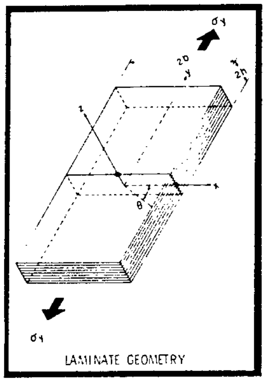
OBJECTIVE

DEVELOP AN ACCURATE AND EFFICIENT MODEL TO DEFINE ELASTIC STRESS FIELDS IN MULTI-LAYERED COMPOSITE LAMINATES, INCLUDING EFFECT OF GEOMETRIC COMPLEXITIES

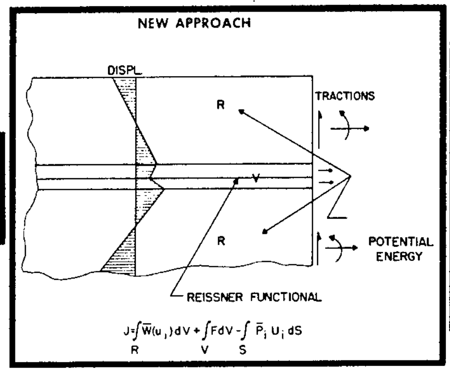
CONCLUSIONS TO DATE

1. GLOBAL-LOCAL MODEL DEVELOPED.
2. VALIDATION VS. FREE-EDGE PROBLEM ACHIEVED (EXP. AND ANALYTICAL)
3. INTERLAMINAR FAILURE MODEL BASED ON AVE. STRESS CORRELATED TO EXPERIMENTAL RESULTS.
4. 2-D CODE COMPLETED.

GLOBAL-LOCAL LAMINATE VARIATIONAL MODEL



1. ELASTICITY - UNTRACTABLE
2. FINITE EL. - IMPRACTICAL
3. APPROX. THEORIES - INACCURATE



FORMULATION OF LAMINATE THEORY

Assumed stress (within each layer)

$$\sigma_x = \frac{1}{h}\left(N_x + \frac{6M_x\zeta}{h}\right)$$

$$\sigma_y = \frac{1}{h}\left(N_y + \frac{6M_y\zeta}{h}\right) \qquad \zeta = \frac{2z}{h}$$

$$\tau_{xy} = \frac{1}{h}\left(N_{xy} + \frac{6M_{xy}\zeta}{h}\right)$$

$$\tau_{xz} = \left(\frac{t_2 - t_1}{2}\right)\zeta + \left(\frac{t_1 + t_2}{4}\right)(3\zeta^2 - 1) + \frac{3V_x}{2h}(1-\zeta^2)$$

$$\tau_{yz} = \left(\frac{s_2 - s_1}{2}\right)\zeta + \left(\frac{s_1 + s_2}{4}\right)(3\zeta^2 - 1) + \frac{3V_y}{2h}(1-\zeta^2)$$

$$\sigma_z = \left(\frac{p_1 + p_2}{4}\right)(3\zeta^2 - 1) + \left(\frac{p_2 - p_1}{4}\right)(5\zeta^3 - 3\zeta) + \frac{3N_z}{2h}(1-\zeta^2) + \frac{15M_z}{h^2}(\zeta - \zeta^3)$$

GLOBAL DISPLACEMENT FIELD

$$u = u(x,y) + z\psi_x(x,y)$$

$$v = v(x,y) + z\psi_y(x,y)$$

$$w = w(x,y) + z\psi_z(x,y) + \frac{z^2}{2}\phi(x,y)$$

LOCAL EQUILIBRIUM EQUATIONS

$$N_{x,x} + N_{xy,y} + t_2 - t_1 = 0$$

$$N_{xy,x} + N_{y,y} + s_2 - s_1 = 0$$

$$V_{x,x} + V_{y,y} + \frac{20M_z}{h^2} + p_1 - p_2 - \frac{h}{6}(t_{1,x} + t_{2,x} + s_{1,y} + s_{2,y}) = 0$$

$$M_{x,x} + M_{xy,y} - V_x + \frac{h}{2}(t_1 + t_2) = 0$$

$$M_{xy,x} + M_{y,y} - V_y + \frac{h}{2}(s_1 + s_2) = 0$$

$$N_z - \frac{h(p_1 + p_2)}{2} - \frac{h^2}{12}(t_{1,x} - t_{2,x} + s_{1,y} - s_{2,y}) = 0$$

$$V_{x,x} + V_{y,y} + \frac{60M_z}{h^2} + 5(p_1 - p_2) - \frac{h}{2}(t_{1,x} + t_{2,x} + s_{1,y} + s_{2,y}) = 0$$

LOCAL CONSTITUTIVE EQUATIONS

$$h\left(\frac{\bar u_{,x}}{2} - \epsilon_x\right) = S_{11}N_x + S_{12}N_y + S_{13}N_z + S_{16}N_{xy}$$

$$h\left(\frac{\bar v_{,y}}{2} - \epsilon_y\right) = S_{12}N_x + S_{22}N_y + S_{23}N_z + S_{26}N_{xy}$$

$$3w - h\epsilon_z = S_{13}N_x + S_{23}N_y + \frac{6}{5}S_{33}N_z + S_{36}N_{xy} - \frac{S_{33}h}{10}(p_1 + p_2)$$

$$h\left(\frac{\bar u_{,y} + \bar v_{,x}}{2} - \epsilon_{xy}\right) = S_{16}N_x + S_{26}N_y + S_{36}N_z + S_{66}N_{xy}$$

$$\frac{h^2}{3}u_{,x} = S_{11}M_x + S_{12}M_y + S_{13}M_z + S_{16}M_{xy}$$

$$\frac{h^2}{4}v_{,y} = S_{12}M_x + S_{22}M_y + S_{23}M_z + S_{26}M_{xy}$$

$$\frac{5h}{4}(3w - \bar w) = S_{13}M_x + S_{23}M_y + \frac{10}{7}S_{33}M_z + S_{36}M_{xy} + \frac{S_{33}h^2}{28}(p_1 - p_2)$$

$$\frac{h^2}{3}(u_{,y} + v_{,x}) = S_{16}M_x + S_{26}M_y + S_{36}M_z + S_{66}M_{xy}$$

$$\frac{3}{4}\left(\bar w_{,y} - w_{,y} + \frac{4v}{h}\right) = \frac{6}{5h}(S_{44}V_y + S_{45}V_x) - \frac{S_{44}}{10}(s_1 + s_2) - \frac{S_{45}}{10}(t_1 + t_2)$$

$$\frac{3}{4}\left(\bar w_{,x} - w_{,x} + \frac{4u}{h}\right) = \frac{6}{5h}(S_{45}V_y + S_{55}V_x) - \frac{S_{45}}{10}(s_1 + s_2) - \frac{S_{55}}{10}(t_1 + t_2)$$

Interface Conditions

a) Continuity (k = 1, 2, ... N-1)

$$t_2^{(k)} = t_1^{(k+1)}$$

$$s_2^{(k)} = s_1^{(k+1)}$$

$$p_2^{(k)} = p_1^{(k+1)}$$

$$\beta_4^{(k)} - S_{44}^{(k)}T_4^{(k)} - S_{45}^{(k)}T_5^{(k)} + \alpha_4^{(k+1)} - S_{44}^{(k+1)}O_4^{(k+1)} - S_{45}^{(k+1)}O_5^{(k+1)} = 0$$

$$\beta_5^{(k)} - S_{45}^{(k)}T_4^{(k)} - S_{55}^{(k)}T_5^{(k)} + \alpha_5^{(k+1)} - S_{45}^{(k+1)}O_4^{(k+1)} - S_{55}^{(k+1)}O_5^{(k+1)} = 0$$

$$\gamma_2^{(k)} - S_{33}^{(k)}R_2^{(k)} + \gamma_1^{(k+1)} - S_{33}^{(k+1)}R_1^{(k+1)} = 0$$

LOCAL EDGE CONDITIONS

$$N_n\bar u_n,\ N_{ns}\bar u_s,\ M_n u_n,\ M_{ns}u_s,\ \left(\frac{3V_n}{h} - \frac{\tau_1 + \tau_2}{2}\right)\bar w,$$

$$(\tau_2 - \tau_1)w,\ \left(\tau_1 + \tau_2 - \frac{2V_n}{h}\right)w$$

Prescribed Tractions and/or Displacements (k = 1, 2, ... N)

$$t_2 = \bar t_2 \quad \text{or} \quad \beta_5 - S_{45}T_4 - S_{55}T_5 = \bar u_2$$

$$s_2 = \bar s_2 \quad \text{or} \quad \beta_4 - S_{44}T_4 - S_{45}T_5 = \bar v_2$$

$$p_2 = \bar p_2 \quad \text{or} \quad \gamma_2 - S_{33}R_2 = \bar w_2$$

$$t_1 = \bar t_1 \quad \text{or} \quad \alpha_5 - S_{45}O_4 - S_{55}O_5 = \bar u_1$$

$$s_1 = \bar s_1 \quad \text{or} \quad \alpha_4 - S_{44}O_4 - S_{45}O_5 = \bar v_1$$

$$p_1 = \bar p_1 \quad \text{or} \quad \gamma_1 - S_{33}R_1 = \bar w_1$$

GLOBAL EQUILIBRIUM EQUATIONS

$$N_{1,x} + N_{6,y} + t_2 - t_1 = 0$$

$$N_{6,x} + N_{2,y} + s_2 - s_1 = 0$$

$$-N_3 + R_{4,y} + R_{5,x} + \frac{H}{2}(p_1 + p_2) = 0$$

$$M_{1,x} + M_{6,y} - V_5 + \frac{H}{2}(t_2 + t_1) = 0$$

$$M_{6,x} + M_{2,y} - V_4 + \frac{H}{2}(s_2 + s_1) = 0$$

$$V_{5,x} + V_{4,y} + p_2 - p_1 = 0$$

$$-M_3 + S_{4,y} + S_{5,x} + \frac{H^2}{8}(p_2 - p_1) = 0$$

GLOBAL CONSTITUTIVE EQUATIONS

$$N_\alpha = \tilde A_{\alpha\beta}\epsilon_\beta + \tilde B_{\alpha\beta}\kappa_\beta - \bar P_\alpha \qquad (\alpha, \beta = 1, 2, 3, 6)$$

$$M_\alpha = \tilde B_{\alpha\beta}\epsilon_\beta + D_{\alpha\beta}\kappa_\beta - \bar O_\alpha$$

$$V_i = \tilde A_{ij}\epsilon_j + \tilde B_{ij}\kappa_j + \frac{1}{2}D_{ij}\beta_j \qquad (i, j = 4, 5)$$

$$R_i = \tilde B_{ij}\epsilon_j + D_{ij}\kappa_j + \frac{1}{2}F_{ij}\beta_j$$

$$S_i = \frac{1}{2}D_{ij}\epsilon_j + \frac{1}{2}F_{ij}\kappa_j + \frac{1}{4}H_{ij}\beta_j$$

GLOBAL BOUNDARY CONDITIONS

EDGE

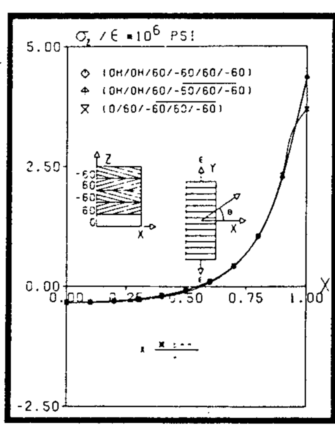
$$N_n u_n,\ N_{ns}u_s,\ M_n\psi_n,\ M_{ns}\psi_s,\ V_n w,\ R_n\psi_z,\ S_n\phi$$

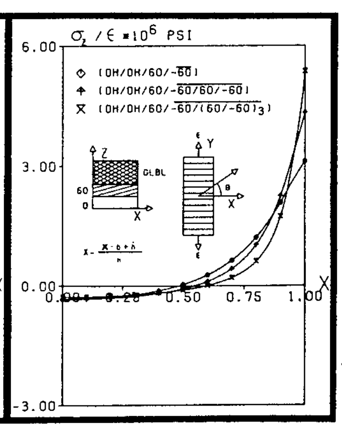
SURFACE

t or $u + h\psi_x$

s or $v + h\psi_y$

p or $w + h\psi_z + \frac{h^2}{2}\phi$





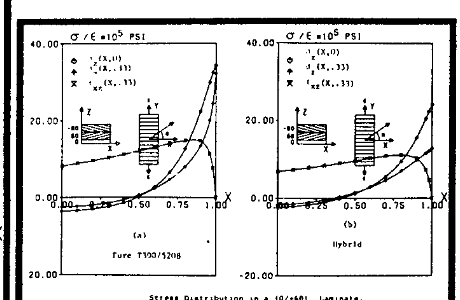


TABLE 5 STRESS AND STRENGTH ANALYSIS AT DELAMINATION

LAMINATE	APPLIED ϵ %	$\bar\sigma_z$	APPLIED LAMINATE σ (KSI)	INPLANE STRENGTHS (KSI)
$(0/90/\pm45)_s$	.52	7.9	40.5	21.7, 39.5, 75.0
$(0/\pm45/90)$	.66	8.1	51.4	21.7, 39.5, 75.0
$(0_2/\pm45_2/90_2)_s$	.54	9.0	42.1	21.7, 39.5, 75.0
$(0/\pm75/90)_s$	.68	7.6	96.0	37.6, 113.0, 121.1
$(0/\pm60/90)_s$	.55	9.2	59.6	29.0, 78.3, 104 2
$(0/\pm60)_s$	.39	8.4	30.4	21.7 56.3
$(0/\pm60_2)_s$	.33	9.0	28.2	22.9, 69.5
$(0/\pm60_4)_s$	.26	8.7	22.4	22.3, 76.1
$(0/\pm60_6)_s$	.22	7.7	18.7	21.3, 76.1
$(\pm30/\pm60)_s$	.61	8.4	34.68	21.7, 46.6
$(\pm30_2/\pm60_2)_s$	.53	8.9	29.75	21.7, 46.6
$(0_2/\pm60_2)_s$	.36	9.2	27.7	21.7, 56.3
$(0_4/\pm60_4)_s$	.25	7.7	19.46	21.7, 56.3
$(0_3/\pm45/90_3)_s$	.35	7.0	27.33	21.7, 39.5, 75.0
$(0_3/\pm45/90)_s$	.67	9.6	40.61	17.5, 28.2, 50.4
$(0_6/\pm45/90)_s$	.54	5.6	24.66	14.1, 21.5, 36.6
$(\pm45/0/90)_s$	-.66	6.6	-51.76	-(53, 121, 151)
$(\pm45/0_3/90)_s$	-.68	8.3	-40.67	-(51, 106, 126)
$(\pm45/0_6/90)_s$	-.78	9.4	-35.37	-(41, 84, 97)

An Iterative Approach for the Evaluation of Delamination
Stresses in Laminated Composites

by Roshdy S. Barsoum

Army Materials & Mechanics Research Center
Watertown, MA 02172

(9 Figures)

OBJECTIVES:

To develop a finite element procedure for the evaluation of delamination stresses in laminated fiber-composites of complex geometries and boundary conditions which are encountered in practical engineering constructions without taxing the computer to its limits.

APPROACH:

A special inter-laminar shear elements is used with an iterative procedurer which uses the classical laminate theory as first approximation to 3-D analysis to obtain the out-of-plane stresses. The Conjugate Gradient Method is used in the iteration scheme.

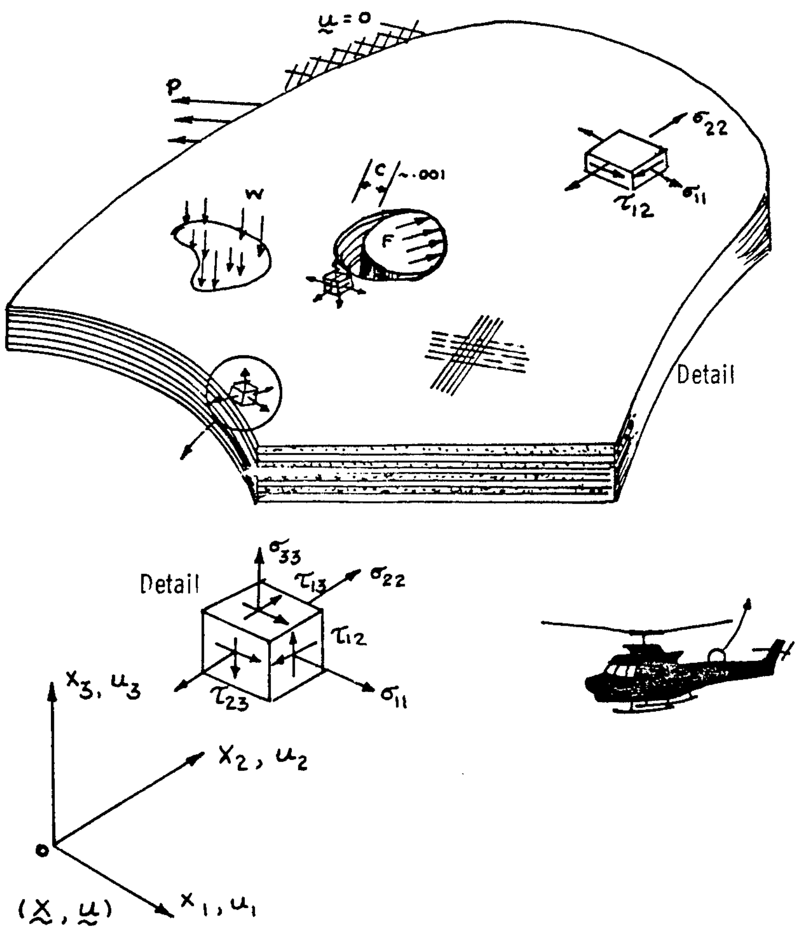


Figure I. Stresses in a general laminated construction

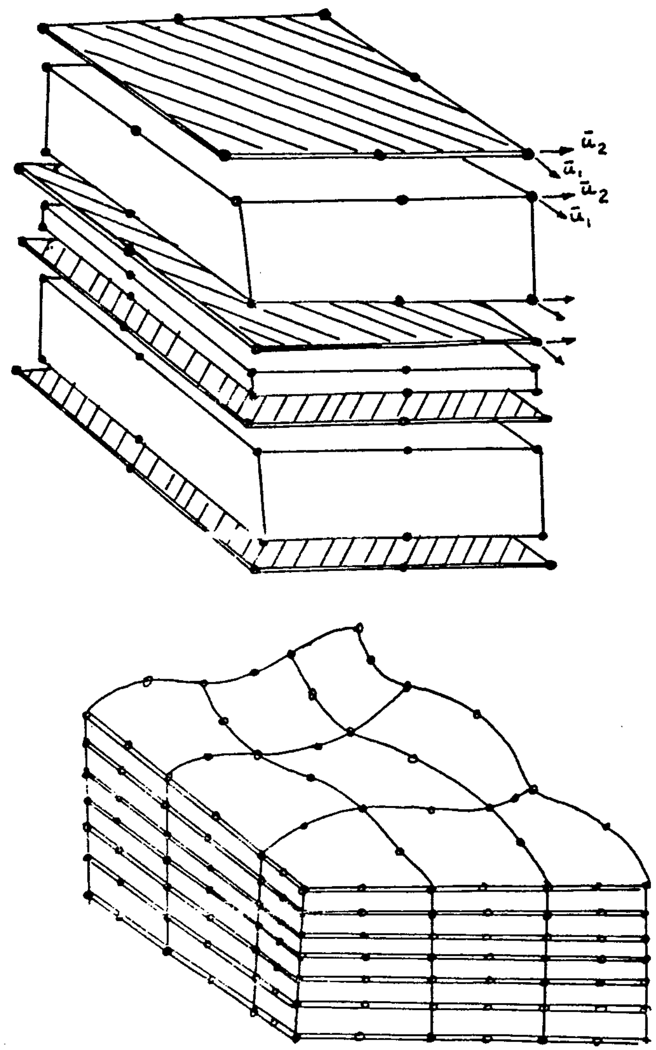
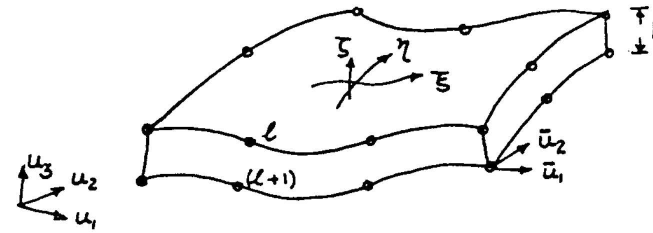
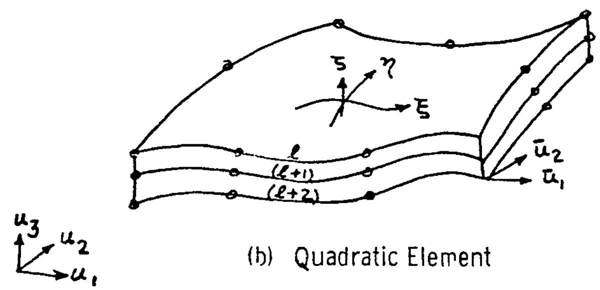


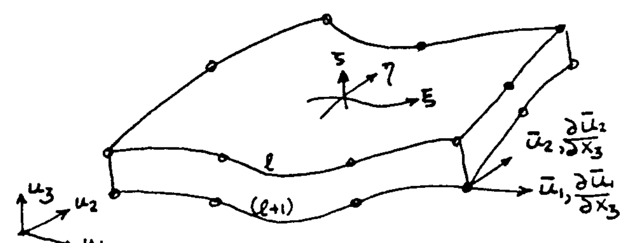
Figure 2. In-Plane Isoparametric and Interlaminar Shear Element Idealization



(a) Linear Element

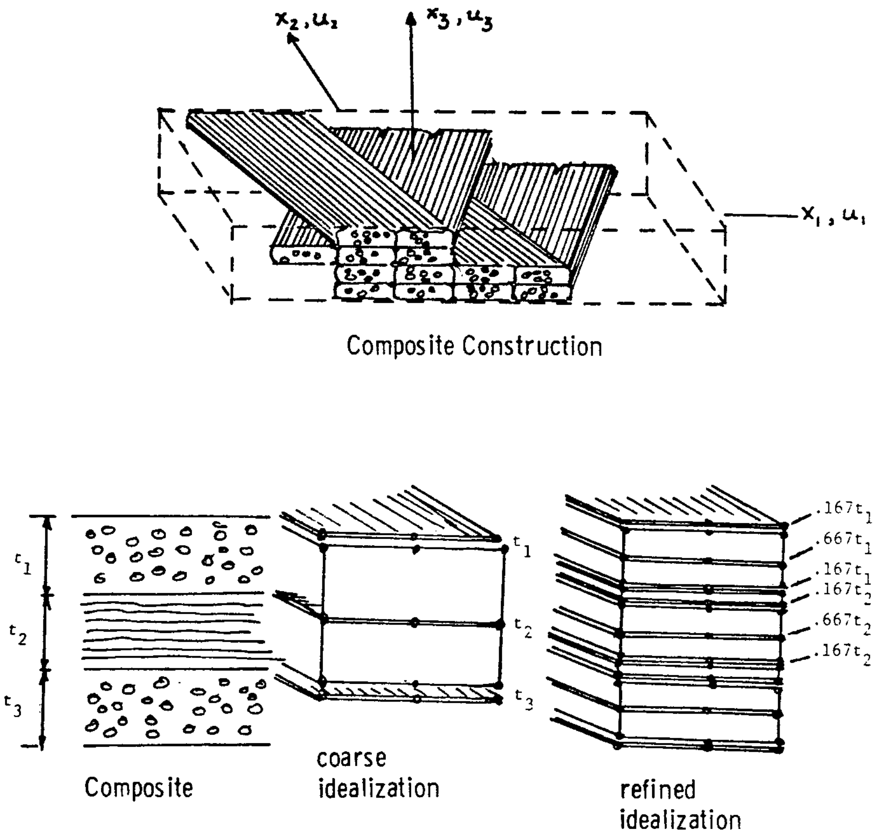


(b) Quadratic Element



(c) Cubic Element

Figure 3. Interlaminar Shear Elements.



Figuer 4. Composite Idealization

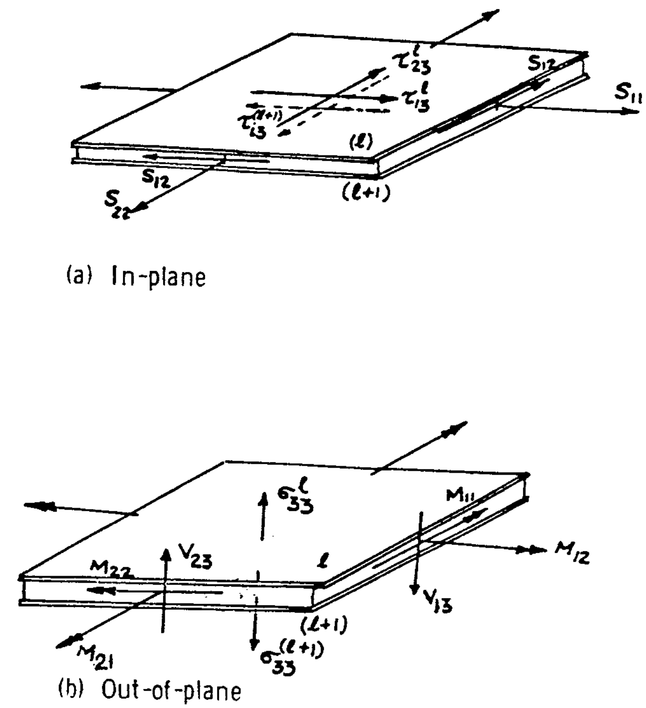


Figure 5. Stress Resultants

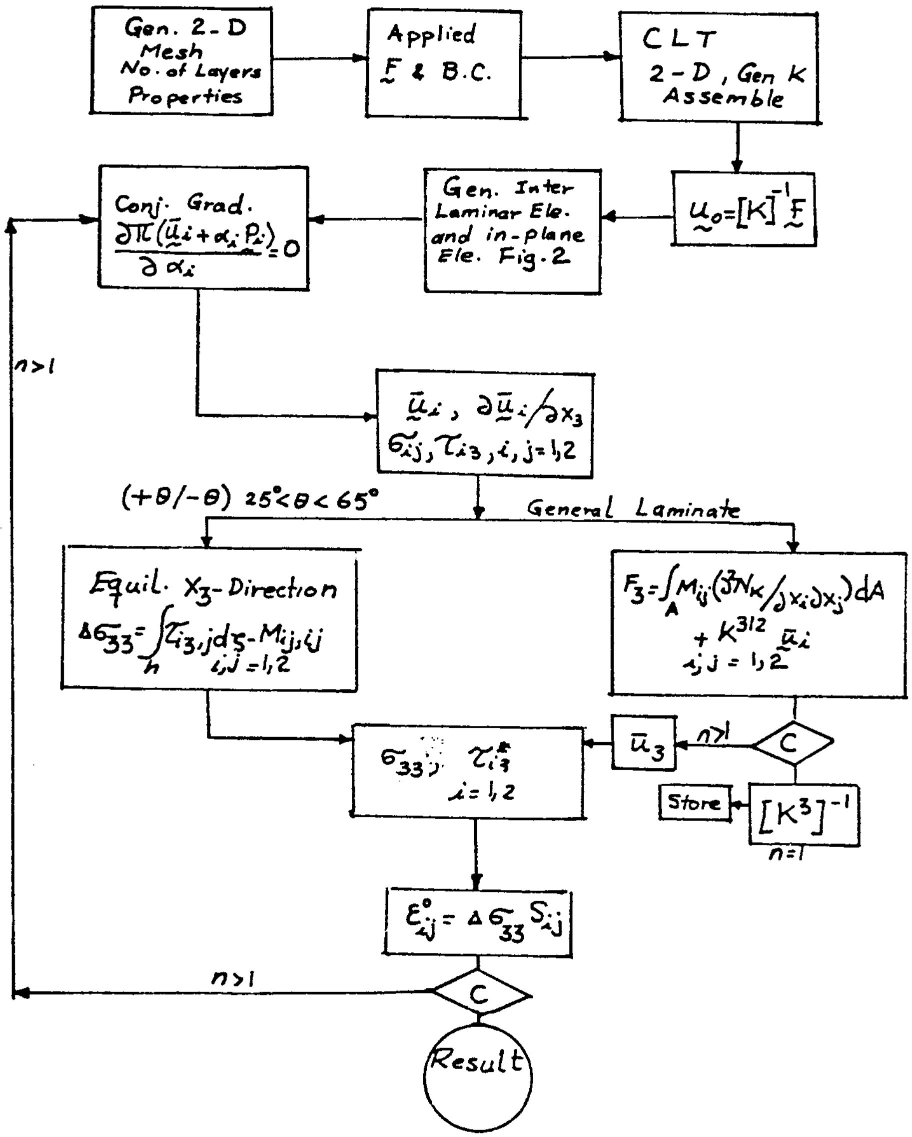
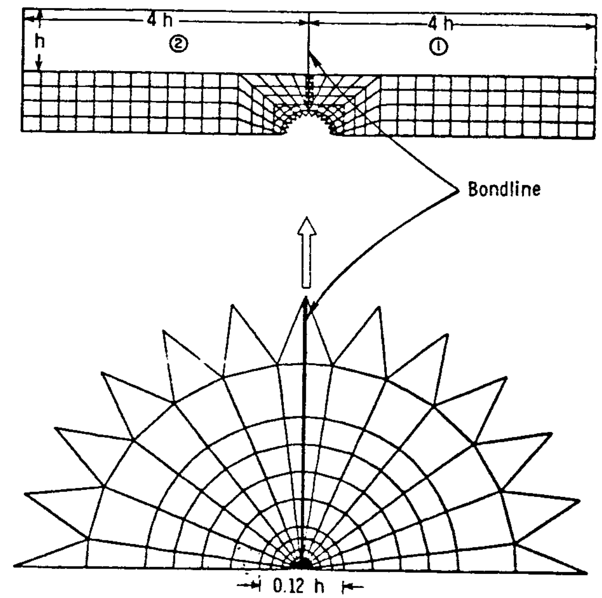


Figure 6. Flow Diagram



FINITE ELEMENT MESH FOR BIMATERIAL TENSION STRIP PROBLEM

Figure 7. Conventional F. E. Idealization

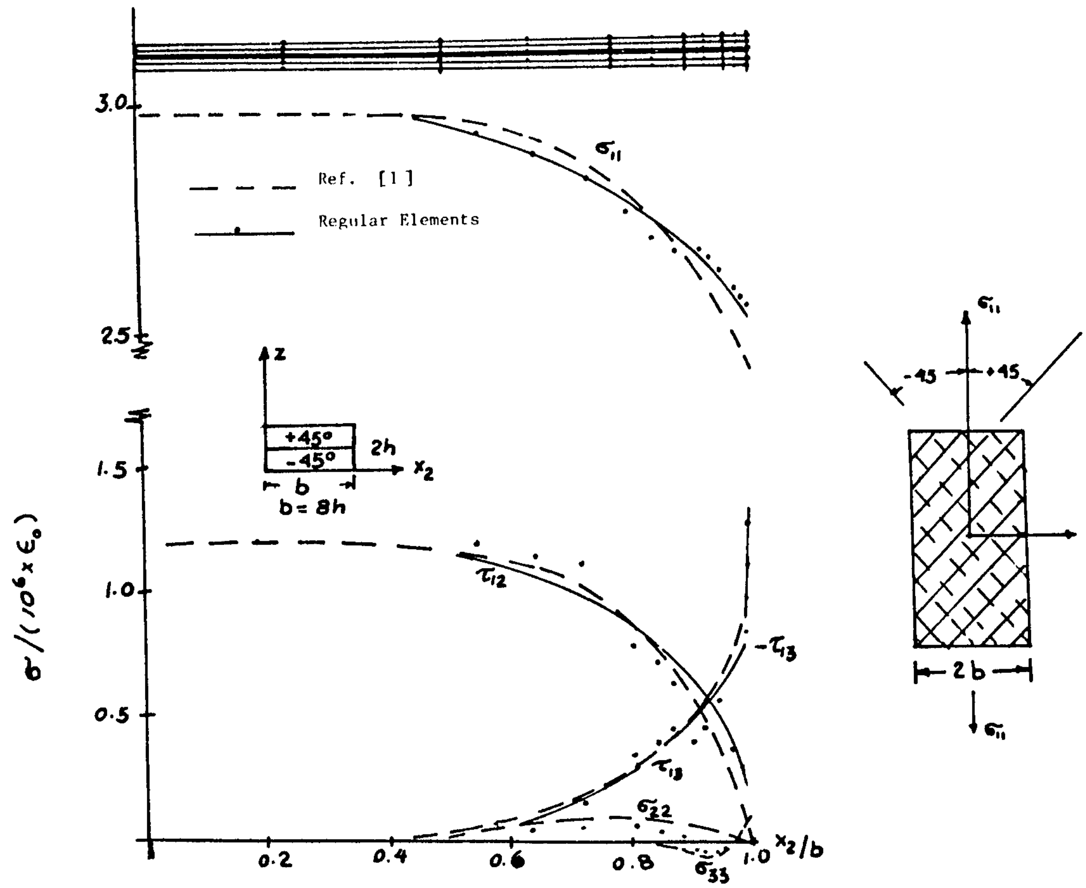


Figure 9. (+45/-45)$_S$ Laminate

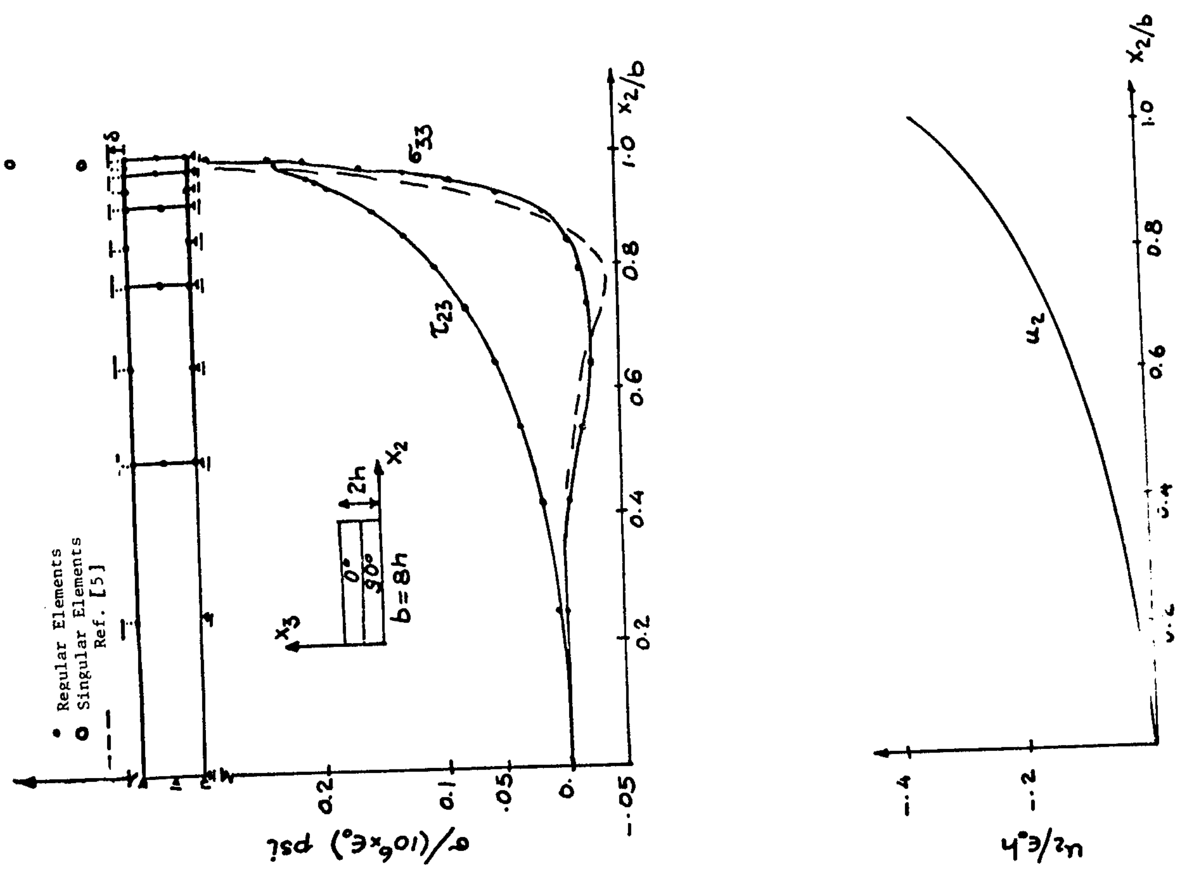


Figure 8 (0/90) Laminate

APPENDIX A: ABSTRACTS

TITLE: DURABILITY OF COMPOSITE STRUCTURE

R. S. Whitehead, G. L. Ritchie, Northrop Corporation

The primary objective of Northrop/AFWAL Composite Wing/Fuselage Program is to demonstrate structural integrity and durability of primary composite structures. In order to achieve this objective, an extensive durability data base has been developed on specimens ranging in complexity from coupons through full-scale wing and fuselage components. Durability test environments ranged from room temperature ambient through real flight time. The test data showed that composites have excellent durability for all environments and specimen geometries. Maximum strength reductions after two lifetimes of severe fatigue loading was 17 percent.

TITLE: DAMAGE TOLERANCE CHARACTERISTICS OF KEVLAR-EPOXY LAMINATES LOADED IN COMPRESSION

J. G. Williams, J. H. Starnes, NASA Langley Research Center, W. Allen Waters, Kentron Technical Center

The damage-tolerance characteristics of Kevlar-epoxy laminates loaded in compression is presented. Kevlar-epoxy laminates with impact damage and with open holes were tested to failure. The results of the tests show that Kevlar-epoxy laminates are susceptible to damage and local discontinuities in a manner similar to that for graphite-epoxy laminates. However, local damage was found to propagate at higher strain levels for certain Kevlar-epoxy laminates than for similar graphite-epoxy laminates. Stiffened panels with Kevlar-epoxy skins and graphite-epoxy stiffeners were also subjected to damage and their damage-tolerance characteristics are also presented. The relative merits of attaching the stiffeners by bolting and bonding was studied and the results of this study are also discussed.

TITLE: COMPRESSION STRENGTH OF COMPOSITES WITH EMBEDDED DELAMINATIONS

R. B. Deo, R. S. Whitehead, M. M. Ratwani, Northrop Corporation

The increasing use of composite materials in military aircraft and the need to ensure their damage tolerance makes it essential that damage tolerance specifications be available for composite structures. The development of damage tolerance requirements for composites is being carried out in an Air Force sponsored program. As a part of this program, an experimental investigation was conducted to determine the influence of embedded delaminations on the strength of graphite/epoxy laminates. The test program was aimed at augmenting the existing data base on the effects of various flaw/damage types to which composite materials are susceptible. This paper presents an assessment of the influence of delaminations on static and fatigue response of composites and the results of the experimental program.

The objective of the test program was to determine the static strength and local buckling response of composite laminates with a lay-up representative of highly loaded wing-skins. The parameters investigated were delamination shape, size, and location, and the influence of stitching in retarding delamination growth. The results show that the responses of small, near-

surface and large, deep delaminations are very similar. In addition, the data show that stitching does not affect delamination growth rate under static loading but is effective in retarding delamination growth under compression fatigue loading.

TITLE: FRACTURE TOUGHNESS OF COMPOSITE LAMINATES
C. C. Poe, Jr., NASA Langley Research Center

Fracture toughness values for composite laminates were predicted using the general fracture toughness parameter and lamina properties. The sensitivity of toughness to layup and fiber and matrix properties was determined. Toughness increased with the strength of fibers in the loading direction and the stiffness of fibers in the off-axis plies. A shear-lag analysis showed that splitting in the 0° plies at the crack tips elevated the toughness of some layups. Otherwise, matrix properties influenced toughness only through lamina elastic constants. The influence of laminate thickness was investigated experimentally. Crack-tip damage was small for the thick laminates and was confined to the plies near the surface. Consequently, for some layups, thick laminates were less tough than thin laminates.

TITLE: PROGRESSIVE FRACTURE OF COMPOSITES
T. B. Irvine, C. A. Ginty, NASA Lewis Research Center

Work-in-progress is described on the Composite Fracture Characterization program being conducted at NASA Lewis Research Center. The purpose of the program is to develop and refine models/procedures for predicting progressive composite fracture. Unique Lewis Research Center capabilities including the Composite Durability Structural Analysis (CODSTRAN) computer code and the Real-Time Ultrasonic C-Scan (RUSCAN) are utilized. Embedded in CODSTRAN are composite mechanics, finite element stress analysis, and failure criteria modules. the RUSCAN facility is used to verify CODSTRAN predicted composite fracture. Results indicate that CODSTRAN/RUSCAN is an effective method of studying progressive fracture of composites.

TITLE: ANALYSIS OF PROGRESSIVE CRACKING IN COMPOSITE MATERIALS
G. J. Dvorak, M. Hejazi, University of Utah, and N. Laws, Cranfield Institute of Technology

This study presents an incremental procedure for evaluation of stiffness changes, matrix crack densities, and local stresses and strains in fiber composite laminae, and in laminated composite plates which are loaded by arbitrary in-plane loads. Stiffness changes in a cracking lamina are found in terms of self-consistent estimates of instantaneous moduli, and the laminate properties are then derived from lamination plate theory. Existing lamina fracture criteria are used to find instantaneous crack densities. Different criteria can be used for polymer and metal matrix systems, and/or quasistatic and

cyclic loading. The analysis is in many ways similar to, and almost as simple as evaluation of elastic properties of fibrous laminates.

TITLE: DAMAGE ACCUMULATION IN COMPOSITES
D. Ulman, General Dynamics/Ft Worth Division

The damage accumulation process in graphite-epoxy ply-termination coupons has been extensively documented. Stiffness change and various nondestructive evaluation techniques were used to study damage growth under a wide variety of test conditions. The effects of stress ratio, stress level, and loading type have been performed using a statistically significant number of tests. Results indicate that damage development is systematic and progressive. All of the test parameters have an effect on the failure process, as is evidence by differing damage patterns and/or damage rates for the different loading conditions.

TITLE: INTERLAMINAR AND INTRALAMINAR FRACTURE GROWTH IN COMPOSITE LAMINATES
A. S. D. Wang, Drexel University

The objectives of this research are (1) to study the physical mechanisms of intralaminar (transverse cracking) and interlaminar (delamination) fracture growth processes in graphite-epoxy laminates; (2) to develop a general method from which analytical models are derived for the individual fracture events and their interacting effects; and (3) to conduct experimental case studies in order to test both the basic methodology and the predictive models.

The presentation will highlight the key points in the methodology development as well as major results obtained in the case studies.

TITLE: A STUDY OF POLYMER MATRIX FATIGUE PROPERTIES
E. M. Odom, D. F. Adams, University of Wyoming

Hercules 3501-6 neat epoxy and Hercules X4001 neat bismaleimide specimens were fabricated and tested, both statically and in cyclic fatigue. Axial tensile and torsional shear loadings were utilized, at room temperature and 88°C. To perform this testing, neat resin casting techniques were developed to produce good quality specimens of these high performance structural polymers in neat (unreinforced) form. Techniques were also developed for gripping specimens during testing. General guidelines to follow for gripping specimens will be presented.

The fatigue data generated for these relatively brittle polymers suggests that there is a knee in the S-N curve. This may be an artifact of the scatter in the data, if it is, work currently in progress should lend information to prove or disprove an existance of a knee in the S-N curve. This work will be presented concurrently with the fatigue data.

Extensive scanning electron microscopy (SEM) was performed on the neat resin fracture surface. The observations suggest that the failures of the

torsion specimen were consistently via tensile mode, characteristic of brittle materials. All fracture surfaces indicated three features. These features included a failure iniation site surrounded by a circular mirror surface. The circular mirror surface was surrounded by a very rough surface. Correlations between the size of the failure iniation site and the static strength and fatigue lives were made. These correlations indicate a direct dependence between fatigue life and static strength to the size of the failure iniation site.

During the course of this study it was noted that various tensile specimen geometries seemed to yield different ultimate tensile strengths. Therefore, a size effects study was performed to determine the tensile strength dependence to the cross sectional area. This testing indicated a dependence. Additionally, for a given cross sectional area, the ultimate strength was found to be dependant on the size of the failure iniation site.

During this study, it was found that a rapid dryout of moisture saturated neat resin specimens could cause specimens to crack. A finite element analysis of this phenomena was conducted. The results of this analysis indicated that the specimen cracking could be correlated to the moisture diffusion coefficient and the moisture level content at full saturation. Details of this analysis will be presented.

TITLE: CHARACERIZATION OF ENVIRONMENTAL EFFECTS ON MECHANICAL RESPONSE OF COMPOSITES

M. Roylance, W. Houghton, E. Pattie, Materials Sciences Corporation

The effect of environmental moisture and service loads on the durability of various Kevlar/epoxy and glass/epoxy composites has been investigated. The extent to which service loads interact with absorbed moisture to accelerate environmental degradation in the systems studied has been assessed.

The composite materials studied include a unidirectionally reinforced Kevlar/TGMDA/DDS epoxy, and both E and S glass DGEBA/DICY epoxy 0/90 crossply.

Absorbed moisture is shown to increase the static strength of the unidirectional Kevlar/epoxy, but does not apparently affect the mechanism of fatigue failure. The glass composites exhibit decreases in strength with increasing moisture absorption, but the S-glass composites lose strength much more rapidly than the E-glass, and in some cases the strength of E-glass composites is superior after the absorption of 1 -1.5% by weight of moisture.

TITLE: CHARACTERIZATION OF RESIN MATRIX COMPOSITES AND THE INFLUENCE OF ENVIRONMENTAL FACTORS ON THEM

S. S. Sternstein, Rensselaer Polytechnic Institute

Small strain dynamic mechanical spectroscopy is used to investigate in-situ resin behavior in high performance composites. Examples of both reversible and irreversible moisture effects are considered. Substantial

interfacial failure is indicated. Preliminary applications of the method to modified (rubber/thermoplastic) epoxy systems and thermoplastic systems are presented; in particular, characterizations of thermal history effects and morphological changes are illustrated. The complex rheological behavior of thermoplastic matrices is demonstrated. Further, delamination studies using the centro-symmetric deformation geometry are also given.

TITLE: RESEARCH ON COMPOSITE MATERIALS FOR STRUCTURAL DESIGN
Y. Weitsman, B. Harper, Texas A&M University

Research on composites at Texas A&M University sponsored by the Air Force Office of Scientific Research is reviewed. Much of the effort is concerned with several student/faculty research projects. The following four studies that are nearing completion or are completed are reviewed: "On the Effects of Post Cure Cool Down and Environmental Conditioning on Residual Stresses in Composite Laminates", (Harper/Weitsman); "Moisture Diffusion in Hybrids", (Clark/Weitsman); "Stress Effects on Moisture Diffusion", (Porth/Weitsman); and "Moisture Induced Damage in Composites", (Jackson/Weitsman).

Several additional investigations, some of which concern modeling of damage in composites, are in progress. These efforts are not reviewed herein.

TITLE: EFFECT OF STRAIN RATE ON GRAPHITE/EPOXY LAMINATES
J. Alper, Naval Air Development Center

The main objective of this program was to experimentally evaluate the effect of strain rate on graphite/epoxy laminates. Specimens representing four laminates ($[0]_{24T}$, $[\pm 45]_{12S}$, $[(\pm 45/0_2)_3/90]_S$, and $[(\pm 45/0/\pm 45)_2 / \pm 45/90]_S$) were statically tested in one of two environments, room temperature dry or elevated temperature wet (200°F, 1% moisture content) at strain rates in either tension or compression which yielded failure times of .1 sec, 1 sec, 10 sec, and 100 sec. Results indicate that the laminates are more strain rate sensitive under the elevated temperature wet environment and that laminates typical for Navy aircraft have a statistically significant strain rate effect only under the elevated temperature wet condition.

TITLE: MIXED MODE FRACTURE OF UNIDIRECTIONAL COMPOSITES
S. L. Donaldson, AFWAL/Materials Laboratory

The results of an in-house program to examine the applicability of the off-axis tensile test to determine the mixed mode fracture characteristics of a unidirectional composite are presented. The off-axis test is shown to produce reliable data from pure Mode I (opening) to $K_I/K_{II} = 0.176$. Pure K_{II} (shearing) values are obtained using a notched, unidirectional

three rail shear test. An empirical relation is given to fit the data. Conversion is then made from critical stress intensity values to critical strain energy release rate values. A brief sensitivity study for this conversion is given. Finally, preliminary SEM views of the fracture surfaces are presented.

TITLE: SUPPRESSION OF DELAMINATION IN COMPOSITES BY THICKNESS DIRECTION REINFORCEMENT

C. T. Sun, Purdue University

Abstract not received in time.

TITLE: ACOUSTIC EMISSION AS AN NDT TOOL FOR COMPOSITES UNDER QUASI-STATIC AND FATIGUE LOADING

J. Awerbuch, Drexel University

Abstract not received in time.

TITLE: MECHANICAL CHARACTERIZATION OF "MAGNAWEAVE" BRAIDED COMPOSITES

L. W. Gause, Naval Air Development Center

The mechanical and impact properties of graphite/epoxy composites manufactured using a general braiding process are being evaluated for possible flight vehicle applications. This new process achieves a fully integrated, three-dimensional orientation of the fibers. Motivating this study is the desire to improve the impact resistance, short-transverse strength and overcome the delamination tendencies of conventional, laminated composites. Two styles of braided test coupons have been fabricated and tested. Results show the braid to have similar strength and elastic properties to corresponding, angle-plied laminates while greatly limiting the extent of impact damage. The braid does not increase the impact damage threshold, however.

TITLE: FRACTURE BEHAVIOR OF CERAMIC COMPOSITES

K. W. Buesking, Materials Sciences Corporation

A combined experimental and analytical study is described which investigated the strength and fracture toughness of whisker reinforced ceramics. Experiments were performed on Al_2O_3 reinforced with SiC whiskers mechanically loaded in four-point flexure. The results showed an increase in flexural strength and K_{IC} as the whisker content of the composites was increased. Several fracture and strength theories were compared to the experimental results. The hypothesis which appeared most consistent with the data treated the composites as though they contained inherent flaws which were the length of the mean free path between reinforcing whiskers. Using this crack size, the measured flexural strength of the composites could be predicted by applying linear elastic fracture mechanics.

TITLE: ANALYTICAL RESULTS FOR POSTBUCKLING BEHAVIOR OF ORTHOTROPIC COMPOSITE PLATES IN COMPRESSION AND IN SHEAR

M. Stein, NASA Langley Research Center

Postbuckling results are presented for long plates loaded in longitudinal compression and in shear. Transverse inplane constraints at the edges of the plate are imposed that might be an upper limit to constraints expected in actual structures and experiment. Comparisons are made between results for plates with transverse displacement constraints and for plates with zero average stress across the width. Both simply supported and clamped edge boundary conditions for out-of-plane deflections are treated. Postbuckling results for compression are insensitive to changes in inplane edge constraints. Results for shear show that changes in inplane edge constraints can cause large changes in postbuckling stiffness.

TITLE: EXPERIMENTAL AND ANALYTICAL STUDIES OF EFFECTS OF NONLINEAR RESPONSE ON THE MECHANICAL PERFORMANCE OF NOTCHED LAMINATES

D. W. Oplinger, C. E. Freese and K. R. Gandhi, Army Materials and Mechanics Research Center

Previous experimental results based on applications of the moire method to pin loaded composite and metallic lugs will be reviewed. Results of these efforts demonstrated that nonlinear behavior plays an important part in the response of 0/90 and ±45 laminates in pinned and bolted joint configurations. Current efforts are aimed at applying nonlinear orthotropic finite element calculations to the pin loaded hole and open-hole situations in such laminates. To this end of finite element program has been developed in which a nonlinear laminate analysis routine is used to provide for constitutive relations. The approach used is iterative, the first step of which is a linear elastic analysis, following which the strains at each Gauss point are fed into the nonlinear laminate analysis to obtain revised estimates of constitutive properties amounting to secant moduli. The process is repeated until "selling out" is indicated by the lack of change in results from one interaction to the next. Calculations obtained to date for ±45 and 0/90 laminates containing open-holes and 0/90 pin loaded laminates will be discussed.

TITLE: BUCKLING OF SURFACE DELAMINATIONS IN QUASI-ISOTROPIC COMPOSITE LAMINATES

K. N. Shivakumar, Old Dominion University and J. D. Whitcomb, NASA Langley Research Center

Buckling of a delaminated region can cause high interlaminar stresses which lead to delamination growth. Hence, the buckling strain is an important parameter in assessing the criticality of the delamination. The objective of this study was to predict the buckling of an elliptic delamination embedded near the surface of a thick quasi-isotropic laminate. The thickness of the delaminated plies group, called the sublaminate, is assumed to be small compared to the total laminate thickness. Finite-element and Rayleigh-Ritz methods were used for the analyses. The Rayleigh-Ritz method was found to be

simple, inexpensive, and accurate, except for highly anisotropic buckled regions. Effects of delamination shape and orientation, material anisotropy, and stacking sequence on buckling strains were examined. Results showed that (1) the stress state around the delaminated region is biaxial, which may lead to buckling when the laminate is loaded in tension, (2) buckling strains for multi-directional fiber sublaminates tend to be bounded by 0° and 90° fiber sublaminates, and (3) delamination growth direction correlates with the direction of elongation of the delamination which yields the lowest buckling strain.

TITLE: APPLICATION OF OPTIMIZATION TECHNIQUES TO COMPOSITE LAMINATES
G. V. Flanagan, AFWAL/Materials Laboratory

A series of highly efficient optimization algorithms have been developed for designing minimum thickness laminates subject to strength constraints with in-plane loads and deflections. These programs are compact and fast enough to run on the smallest microcomputers. Ply ratios, orientations, and orthotropic axis orientation can be optimized, subject to multiple independent loads.

Using these techniques, the potential weight savings over quasi-isotropic laminates, and the number of initial orientations needed were investigated. One finding was that ply ratio optimization, angle optimization, and ply ratio optimization of an orthotropic laminate (with optimal orthotropic axis) will each yield an almost equally efficient laminate.

TITLE: COMPOSITE MECHANICS/RELATED ACTIVITIES AT LEWIS RESEARCH CENTER
C. C. Chamis, NASA Lewis Research Center

Lewis research activities and progress in composite mechanics and closely related areas are summarized. The research activities summarized include: (1) Composite Mechanics; (2) Computer Programs for Composites; (3) High Temperature Composites; and (4) Composite Engine Structural Components. The research activity focus is on. (1) Composite Mechanics -- simplified micromechanics equations, finite element substructuring for composite mechanics and laminate analysis, life/durability and failure modes; (2) Computer Programs for Composites -- intraply hybrid composite design/analysis, integrated composite analysis, structural composite durability, composite thermoviscoplastic structural analysis, and structural tailoring; (3) High Temperature Composites -- test methods development and characterization, composite burner liners, and tungsten-fiber reinforced superalloys (FRS); and (4) Composite Engine Structural Components -- composite frames, fan blades from superhybrid or with composite in-lays, swept turboprops and props for general aviation aircraft, and FRS turbine blades.

TITLE: STATISTICAL EVALUATION OF FAILURE DATA FOR COMPOSITE MATERIAL

D. Neal, L. Spiridgliozzi, Army Materials & Mechanics Research Center

Suggested statistical procedures for obtaining material "A" and "B" allowables from both complete and censored samples are outlined in this paper. The allowables represent a value determined from a specified probability of survival with a 95% confidence in the assertion. The survival probabilities are .99 for the "A" allowables and .90 for the "B" allowables. Both parametric and non-parametric statistical models are evaluated with respect to their desirability in obtaining the allowables. Exploratory data analysis procedures are introduced in order to determine acceptable distribution functions for representing the data in addition to recognizing outliers (bad data) or multi-modality. It is demonstrated from a variety of materials test data that allowable determinations require prior application of exploratory data analysis procedures in order to assure acceptable results. The analysis also provides a process for recognizing either poor testing procedures or inferior material processing.

The two parameter Weibull, normal, lognormal distribution functions are the conventional statistical models for computing the allowables (when non-parametric methods are not applicable). The will usually provide an acceptable range of possible allowable values. The Informative Quantile Function is applied to the test data in order to select the function that best represents the data. In determining the allowables, the desirability of the extreme value function application is shown when limited number of probability ranked data values are available in the primary region (lower ranked numbers) of interest. The required conservatism in this region is satisfied while also satisfying criteria for acceptability of the data representation. The existence of multi-modality or gross outliers in the data set, will in some instances introduce excessively conservative estimates of the allowables when the Weibull function is applied. If multi-modality or outliers are a reality then a suggested procedure using the Breiman Method is used. This method is a recent development for estimating the tail probabilities.

In order to demonstrate the desirability of the methods, allowables have been determined for Kevlar, Graphite, and Glass composite materials subjected to shear, tensile, and compressive loads. Most of the test data was obtained from the MIL-HDBK-17 (USA Army Materials and Mechanics Research Center) project for composite material applications in aircraft structures.

TITLE: GLOBAL-LOCAL MODEL FOR LAMINATE ANALYSIS

N. J. Pagano, AFWAL/Materials Laboratory

The absence of a unified, tractable model to predict the elastic response of a multi-layered laminate (say 100 layers) has foiled attempts to understand the failure modes of practical composite structures. Global models, which follow from an assumed displacement field and lead to the definition of effective (or smeared) laminate moduli, are not sufficiently accurate for stress field computation. On the other hand, local models, in which each layer is represented as a homogeneous anisotropic continuum, become intractable as the number of layers becomes even moderately large (approximately 10). In this work, we blend these concepts into a self-

consistent model which can define detailed response functions in a region of interest (local), while representing the remainder of the domain by effective properties (global). In this investigation the laminate thickness is divided into two parts. A variational principle has been used to derive the governing equations of equilibrium. For the global region of the laminate, potential energy has been utilized, while the Reissner functional has been used for the local region. The field equations are based upon an assumed thickness distribution of stress components within each layer of the local region and displacement components in the global region. The derived boundary of the global region and the prescribed tractions (pointwise in an elasticity sense) satisfy the conditions of vanishing resultant force and moment identically. The same conditions are satisfied in the local region. The stress fields obtained by this formulation compare very well with those obtained by other approaches for laminates with a small number of layers. For large number of layers, internally consistent results are achieved by varying the representation of the global region in the present model.

We shall also conduct studies to define more precisely the range of validity of the global-local model by treating a series of laminate problems which feature interlaminar stresses that are quite small in magnitude, as such stresses pose the most severe challenge to the model. We shall next present an example of the use of the global-local model in the definition of interlaminar normal strength of numerous graphite-epoxy laminates from data generated by Kim (1982). Finally, we shall use these experimental results to show that the most sensitive range of the global-local model is well outside the region of practical interest, at least in the static response of free-edge laminates, by demonstrating that the interlaminar stresses in this range do not appear to exert significant influence on the laminate failure process.

TITLE: AN ITERATIVE APPROACH FOR THE EVALUATION OF DELAMINATION STRESSES IN LAMINATED COMPOSITES

R. Barsoum, Army Materials and Mechanics Research Center

Abstract not received in time.

APPENDIX B: PROGRAM LISTINGS

AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
MATERIALS LABORATORY

INHOUSE

ADVANCED COMPOSITES
WORK UNIT DIRECTIVE (WUD) NUMBER 45
77 April - 85 April

WUD Leader: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Objective: The objective of the current thrust under this work is to develop and demonstrate concepts of damage resistance as applied to fiber reinforced composite laminates. Short term objectives (1-2 yrs) include the following:

(a) Development of failure mode models with emphasis on delamination and matrix cracking.
(b) Assess the role of matrix toughness in composite failure processes.
(c) To develop concepts of interface/interphase strengthening.

CONTRACTS

IMPROVED MATERIALS FOR COMPOSITES AND ADHESIVE JOINTS
F33615-81-C-5056
1 Sept 81 to 31 Aug 84

Project Engineer: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Principal Investigator: Ran Y. Kim
University of Dayton Research Institute
300 College Park Avenue
Dayton, Ohio 45469

Objective: To investigate from both an experimental and analytical standpoint the potential of new and/or modifications of existing materials and reinforcement for use in advanced composite materials and adhesive bonded joints. Such materials are subsequent candidates for use in advanced aircraft and aerospace structural applications.

FAILURE RESISTANT COMPOSITE CONCEPTS—IMPROVED POST-BUCKLING BEHAVIOR
F33615-83-K-5016
1 Jun 83 - 30 Nov 85

Project Engineer: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Principal Investigator: James W. Mar
Technology Laboratory for Advanced Composites
Dept of Aeronautics and Astronautics
Massachusetts Institute of Technology
Cambridge, MA 02139

Objective: The objective of this program is to develop materials concepts for improving the postbuckling behavior of laminated plates and cylindrical shells for application to aircraft structures. Program involves both analytical and experimental work.

CUMULATIVE DAMAGE MODEL FOR COMPOSITE MATERIALS
F33615-80-C-5039
81 Feb 23 - 85 Apr

Project Engineer: Marvin Knight
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-7131 Autovon: 785-7131

Principal Investigator: P. C. Chou
Dyna East Corporation
227 Hemlock Road
Wynnewood, PA 19096
(215) 895-2288

Objective: This program will develop a methodology, including analytical modeling, for predicting and experimentally characterizing advanced composite materials' mechanical responses to defined load histories. A cumulative damage model is the ultimate goal.

CUMULATIVE DAMAGE MODEL FOR COMPOSITE MATERIALS
F33615-81-C-5049
81 Feb 23 - 85 Apr

Project Engineer: Marvin Knight
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-7131 Autovon: 785-7131

Principal Investigator: H. Miller
General Dynamics Corporation
Fort Worth Division
P.O. Box 748
Fort Worth, TX 76101
(817) 732-4811 Ext 5375

Objective: This program will develop a methodology, including analytical modeling, for predicting and experimentally characterizing advanced composite materials' mechanical responses to defined load histories. A cumulative damage model is the ultimate goal.

FUNDAMENTAL MATRIX STIFFNESS FORMULATIONS FOR LAMINATE STRUCTURES
F33615-83-C-5076
1 Jun 83 - 31 Mar 86

Project Engineer: Steven L. Donaldson
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Principal Investigator: Henry T. Yang
School of Aeronautical & Astronautical Engineering
Purdue University
West Lafayette, IN 47907
(317) 494-5117

Objective: This program will develop the mathematical formulation of the stiffness matrices of laminated plates and beams, to ultimately obtain the stress fields, the vibrational, and the buckling response of structural laminates. The elements will include the provision to handle individual failed plies or delaminations. The elements will be formulated in such a way that they can be simply implemented on micro and minicomputers.

CURING PROCESS OF COMPOSITE MATERIALS
F33615-84-(to be awarded)
1 Jan 84 - 1 Oct 87

Project Engineer: Stephen W. Tsai
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-3068 Autovon: 785-3068

Principal Investigator: George S. Springer
Dept of Aeronautics and Astronautics
Stanford University
Stanford, CA 94305

Objective: To extend the analytical modeling developed by the Principal Investigator to include the curing thermosetting and thermal plastics as the matrix materials. To provide criteria for automated process controls and optimization.

AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
FLIGHT DYNAMICS LABORATORY

PROGRAM LISTING

NO INPUT RECEIVED

AIR FORCE OFFICE OF SCIENTIFIC RESEARCH

INHOUSE

NONE

CONTRACTS

BOUNDARY ELEMENTS FOR DEBOND STRESS ANALYSIS
82 March 01 - 83 February 28

Project Engineer: Maj David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Colin Atkinson
Dept of Mathematics
Imperial College of Science & Technology
London SW7 2BZ England

Objective: To develop a boundary integral equation method valid for short crack initiation at the fiber-matrix interface in composite materials.

FRACTURE BEHAVIOR OF BORON ALUMINUM COMPOSITES
79 April 01 - 83 October 14

Project Engineer: Maj David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Jonathan Awerbuch
Dept of Mechanical Engr and Mechanics
Drexel University
Philadelphia, PA 19104
(215) 895-2291

Objective: To provide insight into the fracture mechanisms in boron aluminum composites at room and elevated temperatures through a comprehensive experimental program and correlation of test data with analytical predictions.

IMPROVED CERAMIC FRACTURE BEHAVIOR FOR HIGH TEMPERATURE TURBINE APPLICATIONS
82 April 01 - 83 July 31

Project Engineer: Maj David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Mr Kent W Buesking
Materials Science Corporation
Blue Bell Office Campus, Merion Towle House
Blue Bell, PA 19422
(215) 542-8400

Objective: To identify the failure modes of fiber-reinforced ceramics and establish the theoretical basis for the development of analytical models capable of predicting these modes.

DAMAGE ESTIMATION IN CARBON FIBRE REINFORCED EPOXY AND ITS INFLUENCE ON RESIDUAL PROPERTIES
82 June 15 - 83 June 14

Project Engineer: Maj David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr A R Bunsell
Ecole Nationale Superieure des Mines de Paris
Centre des Materiaux
BP 87
91003 EVRY cedex
France

Objective: To investigate the failure of fibers and the subsequent accumulation of damage in unidirectional carbon fiber reinforced plastics (cfrp) by using the acoustic emission technique, and to extend a recently developed and verified theory of damage accumulation to unidirectional cfrp subjected to cyclic loading.

ANALYSIS OF DAMAGE PROCESSES IN FIBROUS COMPOSITE LAMINATES
82 September 01 - 84 August 31

Project Engineer: Maj David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr George J Dvorak
Dept of Civil Engineering
University of Utah
Salt Lake City, UT 84112
(801) 581-6931

Objective: To conduct a theoretical study of damage accumulation in unnotched fibrous composite laminates caused by distributed internal cracking in individual layers as well as delamination cracks between layers, under monotonic or cyclic mechanical and thermal loads.

THREE-DIMENSIONAL ANISOTROPIC STRESS CONCENTRATIONS
81 December 01 - 83 December 31

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr R A Eubanks
Dept of Civil Engineering
University of Illinois
Champaign, IL 61820
(217) 333-6946

Objective: To develop rigorous analytical methods for three-dimensional stress concentrations in transversely isotropic materials such as advanced composites or other reinforced or layered materials.

FRACTURE, FATIGUE, DYNAMICS, AND AEROELASTICITY OF COMPOSITE STRUCTURES
82 January 01 - 83 December 31

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr James W Mar
Dept of Aeronautics and Astronautics
Massachusetts Institute of Technology
Cambridge, MA 02139
(617) 253-2426

Objective: To derive semi-empirical failure criteria for composite laminates based on an extensive experimental data base on fatigue and fracture, to investigate frequency and modal behavior of unbalanced laminate construction with emphasis on assessing the nonlinear behavior due to large deformations and multi-axis response coupling, and to investigate aeroelastic tailoring effects on flutter and divergence of laminated composite lifting surfaces.

NONLINEAR DYNAMIC RESPONSE OF COMPOSITE ROTOR BLADES
82 September 01 - 84 August 31

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Ozden Ochoa
Dept of Mechanical Engineering
Texas A&M Research Foundation
College Station, TX 77843
(713) 845-2022

Objective: To develop nonlinear finite element models suitable for predicting the structural dynamic response and resulting damage of composite rotor blades under impact and other transient excitations.

NONLINEAR TRANSIENT ANALYSIS OF LAYERED COMPOSITE PLATES AND SHELLS
81 April 01 - 83 June 15

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr J N Reddy
Dept of Engineering Science & Mechanics
Virginia Polytechnic Inst & State University
Blacksburg, VA 24061
(703) 961 6744

Objective: To evaluate the stability and convergence characteristics of penalty-finite elements applied to the dynamic analysis (e.g. low velocity impact) of composite plates and shells, and to evolve a transient analysis capability with greatly improved accuracy, numerical stability and computational efficiency.

INTERLAMINAR FRACTURE TOUGHNESS IN RESIN MATRIX COMPOSITES
83 January 01 - 83 December 31

Project Engineer: Maj David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Lawrence W Rehfield
Dept of Aerospace Engineering
Georgia Institute of Technology
Atlanta, GA 30332
(404) 894-3067

Objective: To develop a Mode II interlaminar fracture coupon and test that can be used in both tension and compression testing, can be analyzed conveniently so that behavior can be readily interpreted, and provides an experimental means for isolating Mode II contributions to fracture.

RESEARCH ON COMPOSITE MATERIALS FOR STRUCTURAL DESIGN
82 January 01 - 83 December 31

Project Engineer: Maj David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Richard A Schapery
Dept of Civil Engineering
Texas A&M University
College Station, TX 77843
(713) 845-7512

Objective: To experimentally identify, study in detail, and model analytically the basic mechanisms of structural response of resin matrix composite materials including studies of micro- and macro-mechanisms of fracture, effects of transient temperature and moisture content, behavior and structure of water in polymers, toughening mechanisms in resins, and theoretical models for deformation and fracture behavior.

EFFECT OF LOCAL MATERIAL IMPERFECTIONS ON BUCKLING OF COMPOSITE STRUCTURAL ELEMENTS
83 June 30 - 84 June 29

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr George J Simitses
Dept of Engineering Science & Mechanics
Georgia Institute of Technology
Atlanta, GA 30332
(404) 894-2770

Objective: To investigate the effects of localized material, geometric, and process imperfections on the buckling characterstics of composite structural elements, and to incorporate them in analytical prediction methods.

INTERLAMINAR AND INTRALAMINAR FRACTURE GROWTH IN COMPOSITE MATERIALS
79 September 01 - 83 September 30

Project Engineer: Maj David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Albert S D Wang
Dept of Mechanical Engineering
Drexel University
Philadelphia, PA 19104
(215) 895-2297

Objective: To develop qualitative understanding of and analytical/computational prediction capability for fracture initiation and propagation processes in composite laminates.

NASA LANGLEY RESEARCH CENTER

INHOUSE

EFFECT OF FOIL TOUGHENING ON IMPACT RESISTANCE OF LAMINATES
81 May 1 - 83 November 30

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To determine the effect on impact resistance of partial interlaminar separations between layers of a laminate. Perforated mylar foil produces the partial separations.

MECHANICS OF LOW-VELOCITY IMPACT
81 June 1 - 84 June 30

Project Engineer: Dr. Wolf Elber
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3046 FTS 928-3046

Objective: From quasi-static deformation analysis, determine the criteria for low-velocity impact damage; establish threshold levels for impact damage. Develop fracture mechanics analyses for delamination growth and membrane failure.

NONLINEAR ACOUSTIC ANALYSIS OF COMPOSITES
83 July 1 - 85 June 30

Project Engineer: Dr. William P. Winfree
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3036 FTS 928-3036

Objective: To develop systems for the study of the nonlinear properties of graphite/epoxy composites and related materials to determine damage mechanisms.

SYNTHESIS OF TOUGHENED MATRIX RESIN SYSTEMS
81 October 1 - 85 September 30

Project Engineers: Dr. Terry L. St. Clair
Dr. Vernon L. Bell, Jr.
Paul M. Hergenrother
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Objective: New polymer compositions are being synthesized for evaluation as new toughened graphite composite matrix materials. Linear, thermoplastic polyimides, lightly crosslinked polysulfones, and polyesters, as well as semicrystalline polyesters, are being investigated.

TOUGHNESS TEST METHODOLOGY
80 October 1 - 85 September 30

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Objective: To investigate, develop (if necessary), and select appropriate test methods for screening the impact resistance and fracture toughness properties of neat polymers and composites. Methodology will help guide programs to synthesize new toughened matrix resins. Edge-delamination, double-cantilever-beam, and compact-tension tests are being emphasized using a variety of tough and brittle matrix resins.

FRACTURE OF LAMINATED COUPONS
78 October 1 - 84 September 30

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To develop a methodology to predict residual strengths of damaged composite laminates using, as starting points, lamina properties or possibly the properties of the fibers and matrix. To determine the parameters that lead to tough composites.

DAMAGE TOLERANT COMPOSITE STRUCTURES
74 June 1 - 84 May 31

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To measure the ability of buffer strips and bonded stringers to increase the residual tension strength of damaged panels, and to develop an analysis to predict residual strength in terms of panel configuration and damage size.

EFFECT OF ELEVATED TEMPERATURE ON LARGE GRAPHITE/POLYIMIDE BUFFER STRIP PANELS
81 February 11 - 84 May 31

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To experimentally determine the effect of elevated temperature on the fracture behavior of large graphite/polyimide buffer strip panels with various size buffer strips.

FRACTURE BEHAVIOR OF THICK LAMINATES
81 October 1 - 84 September 30

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To identify potential fracture problems associated with scale-up of graphite/epoxy laminates to thicknesses of about 100 plies and to compare fracture toughness obtained from tests on center-crack, compact-tension, and bending specimens. Both through and part-through thickness slits will be considered.

EFFECT OF MOISTURE AND ELEVATED TEMPERATURE ON GRAPHITE/EPOXY BUFFER STRIP PANELS
80 November 1 - 84 May 31

Project Engineer: C. A. Bigelow
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3191 FTS 928-3191

Objective: To experimentally determine the effect of moisture and elevated temperature on the fatigue life of graphite/epoxy buffer strip panels.

WOVEN COMPOSITE BUFFER STRIP PANELS
81 January 1 - 84 May 31

Project Engineer: John M. Kennedy
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3191 FTS 928-3191

Objective: To demonstrate that buffer strip panels built with woven cloth have the crack-arresting capability of panels built with conventional prepreg tape. Damaged panels will be tested in shear and tension.

STRESS ANALYSIS OF BEARING-LOADED LAMINATES
83 October 1 - 84 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Objective: To calculate stresses near loaded holes in finite-length laminates with tension-reacted and compression-reacted bearing.

MICROMECHANICS ANALYSIS OF INTERLAMINAR FRACTURE
83 July 1 - 86 October 1

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Objective: To develop a basic understanding of the mechanics that govern interlaminar fracture toughness and to develop a neat-resin fracture test that correlates with interlaminar fracture toughness.

FAILURE ANALYSIS OF BEARING-LOADED LAMINATES
82 October 1 - 85 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Objective: To develop the basic understanding of the failure micromechanics that govern damage onset, strength, and fracture toughness for bearing-loading laminates.

ADHESIVE DEBOND CHARACTERIZATION
76 October 1 - 86 September 30

Project Engineers: Dr. W. Steven Johnson
Richard A. Everett, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 928-2715

Objective: To verify that identical specimens manufactured at different facilities using the same adhesive/adherent (7075 Al/FM-73) bonding techniques behave in a similar manner when subjected to cyclic loading. To develop an approach to calculate cyclic debond threshold and rate such that the cyclic behavior of the bondline can be predicted for any geometry (using finite elements) for a given adhesive/adherent system. To expand from metal-to-metal to composite-to-composite bonds and to examine temperature, moisture, and spectrum loading effects.

STRESS ANALYSIS OF ADHESIVE BONDS
80 October 1 - 84 September 30

Project Engineers: Richard A. Everett, Jr.
John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Objective: To review currently available finite-element routines and their applicability to the adhesive bondline stress analysis. To modify available model or develop a new model to assess G_I and G_{II} at debond front, and to incorporate into model material and geometric nonlinear behavior.

FAILURE MODES OF ADHESIVELY BONDED COMPOSITE JOINTS
81 June 1 - 84 September 30

Project Engineers: Dr. W. Steven Johnson
Dr. P. D. Mangalgiri
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Objective: To conduct experimental tests to determine the failure modes and mechanisms of adhesively bonded composite joints. To assess secondary bonding versus co-curing in graphite/epoxy and Kevlar/epoxy joints. Correlate debond growth rates with strain-energy-release rates. Establish design guideline for adhesively bonded composite joints.

REALISTIC ADHESIVELY BONDED JOINT ELEMENT
81 October 1 - 84 September 30

Project Engineer: Richard A. Everett, Jr.
Mail Stop 188E
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Objective: To manufacture several variations of a simple adhesively bonded wing splice joint under contract (metal-to-composite specimens). To determine fatigue and fracture failure modes for a "realistic" aircraft adhesively bonded structure. These joints will consist of titanium wing ribs embedded in graphite/epoxy wing skins.

ADHESIVE BOND CHARACTERIZATION
82 October 1 - 85 September 30

Project Engineer: Carl E. Rucker
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3047 FTS 928-3047

Objective: To measure adhesive mechanical properties in the bonded condition. Develop techniques and assess reliability. NDI will be used to investigate complex modulus and classify relative strength characteristics.

PREDICTION OF FATIGUE LIFE OF NOTCHED COMPOSITE LAMINATES
73 June 1 - 85 September 30

Project Engineers: Dr. T. Kevin O'Brien
John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Objective: To develop a method to design fatigue resistant composite laminates. The method addresses three areas: failure mechanisms are identified; analyses to predict inplane and interlaminar damage growth are developed; and inplane and interlaminar data bases are developed to evaluate the methodology.

PREDICTION OF INSTABILITY-RELATED DELAMINATION GROWTH
79 January 2 - 84 August 31

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Objective: To predict rate of instability-related delamination growth. Approximate stress analyses will be developed based on understanding gained from rigorous analyses. Experiments will be performed to obtain a data base for use by the analysis in making predictions and for verifying and improving the analysis.

PREDICTION OF STIFFNESS LOSS, RESIDUAL STRENGTH, AND FATIGUE LIFE OF UNNOTCHED LAMINATES
80 June 1 - 83 October 31

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Objective: To predict the stiffness loss, residual strength, and fatigue life of realistic unnotched laminates using baseline data from simple laminates.

DETERMINATION OF EFFECT OF RESIN TOUGHNESS ON MECHANICS OF COMPRESSION FAILURE
83 May 1 - 84 August 31

Project Engineers: John D. Whitcomb
Dr. Norman J. Johnston
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Objective: To identify parameters related to the mechanics of compression failure. To develop an analytical model to predict compression failure.

THE EVALUATION OF GRAPHITE/POLYIMIDE HONEYCOMB SANDWICH PANELS
79 June 15 - 83 September 30

Project Engineer: Jane A. Hagaman
Mail Stop 364
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2486 FTS 928-2486

Objective: To evaluate the shear behavior of an optimized sandwich panel at room and elevated temperatures using a diagonal tension test method, and to correlate the behavior with analytical predictions.

FLIGHT SERVICE EVALUATION OF COMPOSITE COMPONENTS ON COMMERCIAL AND MILITARY AIRCRAFT
72 March 1 - 90 December 31

Project Engineer: H. Benson Dexter
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2869 FTS 928-2869

Objective: To evaluate the long-term durability of composite components installed on commercial and military transport and helicopter aircraft. Over 300 components constructed of boron, graphite, and Kevlar composites will be evaluated after extended service. Components include graphite/epoxy rudders, spoilers, tail rotors, vertical stabilizers, Kevlar/epoxy fairings, doors and ramp skins, and boron/aluminum aft pylon skins. Note: Over 2.8 million total component flight hours have been accumulated since initiation of flight service in 1972. Composite components on L-1011, B-737, and DC-10 aircraft have accumulated over 25,000 flight hours each. Excellent in-service performance and maintenance experience have been achieved with the composite components.

POSTBUCKLING RESPONSE OF COMPOSITE MATERIAL SUBJECTED TO SHEAR LOADING
79 July 1 - 85 June 30

Project Engineer: Gary L. Farley
Mail Stop 188A
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2850 FTS 928-2850

Objective: To determine the postbuckling strength of Kevlar and Kevlar-graphite/epoxy composites under static shear and spectrum fatigue loading, as well as low-velocity and ballistic impact. This study will establish a basis for demonstrating the use of thin composite laminates beyond the point of initial shear instability. A shear fixture has been developed that virtually eliminates the adverse stresses in the corners of the shear panel.

THE ENERGY ABSORPTION OF COMPOSITE CRASHWORTHY STRUCTURE
80 August 1 - 85 December 31

Project Engineer: Gary L. Farley
Mail Stop 188A
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2850 FTS 928-2850

Objective: To determine the energy absorption characteristics of glass, Kevlar, and graphite/epoxy composites and to develop the analytical capability to predict the energy absorption characteristics of new composite materials. Tube specimens are being subjected to static and dynamic crushing tests. The research is focused on development of the capability to design efficient crashworthy composite structures for rotorcraft.

ADVANCED CONCEPTS FOR COMPOSITE HELICOPTER FUSELAGE STRUCTURES
83 April 1 - 84 December 31

Project Engineer: Donald J. Baker
Mail Stop 188A
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2850 FTS 928-2850

Objective: Investigate new design concepts for composite materials on lightly loaded helicopter fuselage structures. Trade studies will be performed using the computer code PASCO. Initial studies will be for compression loading. After testing some of the designs for panel compression, trade studies will be performed for combined compression/shear-loaded panels.

EFFECTS OF THERMAL CYCLING ON DIMENSIONAL STABILITY OF GRAPHITE/EPOXY COMPOSITES
81 October 1 - 84 September 30

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2143 FTS 928-2143

Objective: To determine the effects of thermal cycling from 117K to 400K on dimensional stability of graphite/epoxy composites.

DIMENSIONAL STABILITY OF METAL-MATRIX COMPOSITES IN THE SPACE ENVIRONMENT
82 October 1 - 85 September 30

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2143 FTS 928-2143

Objective: To determine and predict the dimensional changes induced by long-time exposure to the space environment.

RADIATION EFFECTS ON MATERIALS FOR STRUCTURAL COMPOSITES
79 July 1 - 84 June 30

Project Engineer: Dr. Edward R. Long, Jr.
Mail Stop 399
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3892 FTS 928-3892

Objective: To determine and correlate the effects of particulate radiation exposure on the properties and chemical structure of materials for structural composites and to develop procedures for accelerated laboratory simulation of long-term missions in a space radiation environment.

EFFECT OF MICROCRACKING ON THE DIMENSIONAL STABILITY OF COMPOSITES
80 October 1 - 84 September 30

Project Engineer: David E. Bowles
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2143 FTS 928-2143

Objective: To develop analytical methods to predict the effect of microcracking on the dimensional stability of graphite/resin composites and correlate with experimental data.

POSTBUCKLING AND CRIPPLING OF COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS
79 March 1 - 84 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Objective: To study the postbuckling and crippling of compression-loaded composite components and to determine the limitations of postbuckling design concepts to structural applications.

DESIGN TECHNOLOGY FOR STIFFENED CURVED COMPOSITE PANELS
79 October 1 - 84 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Objective: To develop verified design technology for generic advanced-composite stiffened curved panels.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH CUTOUTS
77 October 1 - 84 September 30

Project Engineer: Mark Shuart
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2813 FTS 928-2813

Objective: To study the effects of cutouts on the compression strength of composite structural components and to identify the failure modes that govern the behavior of compression-loaded components with cutouts.

POSTBUCKLING OF FLAT STIFFENED GRAPHITE/EPOXY SHEAR WEBS
81 July 1 - 84 September 30

Project Engineer: Marshall Rouse
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4585 FTS 928-4585

Objective: To study the postbuckling response and failure characteristics of flat stiffened graphite/epoxy shear webs.

CURVED GRAPHITE/EPOXY PANELS SUBJECTED TO INTERNAL PRESSURE
80 October 1 - 84 September 30

Project Engineer: Richard L. Boitnott
Mail Stop 190
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3795 FTS 928-3795

Objective: To study the effects of internal pressure on the nonlinear response and failure characteristics of curved graphite/epoxy panels.

POSTBUCKLING ANALYSIS OF GRAPHITE/EPOXY LAMINATES
80 October 1 - 84 September 30

Project Engineer: Dr. Manuel Stein
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2813 FTS 928-2813

Objective: To develop accurate analyses for the postbuckling response of graphite/epoxy laminates and to determine the parameters that govern postbuckling behavior.

STRUCTURAL PANEL ANALYSIS AND SIZING CODE FOR STIFFENED PANELS
79 October 1 - 84 September 30

Project Engineers: Dr. Melvin S. Anderson
Dr. W. Jefferson Stroud
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3054 FTS 928-3054

Objective: To develop an accurate analysis and structural optimization capability for stiffened composite panels subjected to inplane tension, compression, shear, normal pressure, and thermal loads.

CRASH CHARACTERISTICS OF COMPOSITE FUSELAGE STRUCTURE
82 July 1 - 84 September 30

Project Engineer: Huey D. Carden
Mail Stop 495
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3795 FTS 928-3795

Objective: To study the crash characteristics of composite transport fuselage structural components.

DAMAGE TOLERANT DESIGN TECHNOLOGY FOR COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS
78 October 1 - 84 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3524 FTS 928-3524

Objective: To develop structurally efficient design concepts for containing and resisting damage in compression-loaded composite structural components.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH DAMAGE AND LOCAL DISCONTINUITIES
76 October 1 - 84 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3524 FTS 928-3524

Objective: To study the effects of impact damage and local discontinuities on the compression strength of composite structural components, to identify the failure modes that govern the behavior of compression-loaded components subjected to low-velocity impact damage, and to analytically predict failure and structural response.

CONTRACTS

IMPACT CONTACT STRESS ANALYSIS
NAG-1-222
81 November 1 - 83 December 31

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. C. T. Sun
School of Aeronautics and Astronautics
Purdue University
West Lafayette, Indiana 47907
(317) 494-5130

Objective: To integrate the contact behavior and dynamic structural response to solve impact problems involving laminates under initial stress. With the aid of the previously-developed contact law, the dynamic response of the laminate will be modeled by finite elements. Impact damage will be investigated experimentally and correlated with the results of the analysis.

EXPERIMENTAL STUDIES OF IMPACT DAMAGE IN COMPOSITE LAMINATES
NAG-1-366
83 May 17 - 84 May 16

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. I. M. Daniel
Department of Mechanical Engineering
Illinois Institute of Technology
Chicago, Illinois 60616
(312) 567-3185

Objective: To characterize impact damage in graphite/epoxy composite laminates and correlate it with transient strain and deformation history during impact. Plate and beam specimens containing embedded strain gages will be impacted with projectiles of various radii at two velocities.

QUANTITATIVE RECONSTRUCTIVE ULTRASONIC AND THERMAL IMAGING
NCC1-50
83 January 1 - 84 December 31

Project Engineer: Dr. Joseph S. Heyman
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3418 FTS 928-3418

Principal Investigator: Dr. Chris Welch
Virginia Associated Research Campus
College of William and Mary
12070 Jefferson Avenue
Newport News, Virginia 23606
(804) 877-9231

Objective: To develop a state-of-the-art ultrasonic and thermal diffusivity reconstructive imaging system for quantitative materials characterization.

QUANTITATIVE PHYSICAL ANALYSIS OF IMPACT DAMAGE
NSG-1601
80 March 1 - 84 February 29

Project Engineer: Dr. Joseph S. Heyman
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3418 FTS 928-3418

Principal Investigator: Professor James G. Miller
Laboratory for Ultrasonics
Physics Department
Washington University
St. Louis, Missouri 63130
(314) 889-6229

Objective: To improve nondestructive acoustic/ultrasonic techniques for quantitative characterization of defects in composite materials and to investigate new quantitative measurement phenomena applicable to graphite/epoxy.

NEAT RESIN-COMPOSITE PROPERTY RELATIONSHIPS
NAG-1-277
82 May 5 - 84 May 3

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Principal Investigator: Dr. Donald F. Adams
Department of Mechanical Engineering
University of Wyoming
Laramie, Wyoming 82071
(307) 766-2371

Objective: A detailed evaluation of candidate toughened neat resin systems is being conducted, including determination of tensile modulus, tensile strength, Poisson's ratio, shear modulus, shear strength, coefficient of thermal expansion, coefficient of moisture expansion, and strain-energy-release rates. These data will be used along with appropriate micromechanics models to predict expected composite response. These predictions will be compared with composite test results to determine the validity of the model and the influence of neat resin property variations on composite response.

MECHANICAL PROPERTY STUDIES IN HIGH PERFORMANCE COMPOSITES
NAG-1-253
82 January 25 - 84 August 30

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Principal Investigator: Dr. S. S. Sternstein
Department of Materials Engineering
Rensselaer Polytechnic Institute
Troy, New York 12181
(518) 266-6499

Objective: Develop quantitative relationships between neat resin viscoelastic properties and in situ composite resin properties. Determine what effect resin viscoelasticity has on composite mechanical properties, particularly out-of-plane properties. Dynamic mechanical spectroscopic studies are being run using both the three-point and centro-symmetric deformation geometries. Creep, stress relaxation, and biaxial studies are also planned.

DOUBLE-CANTILEVER-BEAM TEST METHOD DEVELOPMENT
L-31134B/NAS1-17074
82 February 1 - 85 January 31

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Principal Investigators: Dr. Don L. Hunston
National Bureau of Standards
Polymer Division, Building 224
Washington, DC 20234
(801) 921-3318

and

Dr. W. D. Bascom
Hercules, Inc.
Aerospace Division
Bacchus Works
Magna, Utah 84044
(801) 250-5911, ext. 3379

Objective: Develop test methods for the interlaminar fracture toughness of composite materials, with particular emphasis on the double-cantilever beam. Specimen geometry (thickness, width, and taper), stacking sequence, rate of fracture, and effects of temperature and humidity will be investigated.

DEVELOPMENT OF HETEROGENEOUS LAMINATING RESINS
NAS1-16798
81 September 30 - 84 February 15

Project Engineer: Paul M. Hergenrother
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Principal Investigator: Larry Hopper
Narmco Materials, Inc.
1440 N. Kraemer Blvd.
Anaheim, California 92806
(714) 630-9400

Objective: Develop technology leading to toughened 350°F cure heterogeneous matrix resins for graphite composites. Target applications involve commercial aircraft transports with -65°F to 200°F temperature range. Compositions involving the addition of 4 to 6 percent thermoplastics to epoxy and epoxy-bismaleimide compositions will be investigated.

DEVELOPMENT OF IMPACT/SOLVENT-RESISTANT THERMOPLASTIC MATRICES
NAS1-16808
81 September 4 - 84 December 31

Project Engineer: Paul M. Hergenrother
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Principal Investigator: Chad B. Delano
Acurex Corporation
Aero-Therm Division
485 Clyde Avenue
Mountain View, California 94042
(415) 964-3200, ext. 3820

Objective: Candidate aliphatic-aromatic heterocyles are being synthesized to develop an impact-and-solvent resistant thermoplastic with acceptable processability in the 600°F range. Heteroaromatics being investigated include polyimides, N-arylenepolybenzimidazoles, and polybenzimidazoles containing both rigid and soft segments.

FRACTURE AND CRACK GROWTH IN ORTHOTROPIC LAMINATES
NSG-1606
79 July 1 - 83 December 31

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. Jonathan Awerbuch
Department of Mechanical Engineering
Drexel University
Philadelphia, Pennsylvania 19104
(215) 895-2291

Objective: To explore the fracture characteristics of graphite/polyimide composites at elevated temperatures using laminates with slits.

FRACTURE AND CRACK GROWTH IN ORTHOTROPIC LAMINATES
NSG-1297
74 October 16 - 84 October 15

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. James G. Goree
Department of Mechanical Engineering
Clemson University
Clemson, South Carolina 29631
(803) 656-3291

Objective: To develop analyses that predict strength of buffer strip panels using models that treat the fiber and matrix as discrete elements.

THE VISCOELASTIC CHARACTERIZATION AND LIFETIME PREDICTION OF STRUCTURAL ADHESIVES
NAG-1-227
81 November 1 - 84 October 31

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Principal Investigator: Dr. H. F. Brinson
Department of Engineering Science and Mechanics
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-6627

Objective: To develop a procedure to predict the failure of adhesive joints where service life must span 10 to 20 years using, as a bais, analytical projections or extrapolations from short-time test data.

MATERIAL CHARACTERIZATION OF STRUCTURAL ADHESIVES IN THE LAP-SHEAR MODE
NAG-1-284
82 June 1 - 84 September 30

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Principal Investigator: Dr. Erol Sancaktar
Mechanical and Industrial Engineering Department
Clarkson College of Technology
Potsdam, New York 13676
(315) 268-2308

Objective: A general method for characterizing structural adhesives in the bonded lap-shear mode is proposed. Two approaches in the form of semi-empirical and theoretical approaches will be used.

FRACTURE AND FATIGUE MECHANISM OF ADHESIVELY BONDED JOINTS
(Grant Pending)
83 October 1 - 84 September 30

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Principal Investigator: Dr. Shankar Mall
Department of Engineering Mechanics
University of Missouri-Rolla
Rolla, Missouri 65401
(314) 341-4599

Objective: Develop a further understanding of adhesively bonded joints by conducting debond studies at different stress ratios and developing fracture toughness data on new adhesive systems.

FATIGUE CRACK GROWTH IN ADHESIVE JOINTS UNDER MODE I-III LOADING
(Contract Pending)
83 October 1 - 84 September 30

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Principal Investigator: Dr. E. J. Ripling
Materials Research Laboratory, Inc.
One Science Road
Glenwood, Illinois 60425
(312) 755-8760

Objective: To develop debond growth rate data under mixed-mode I and III loading. These data will be compared with the mode I and mixed-mode I and II data developed in-house.

FATIGUE DAMAGE IN NOTCHED COMPOSITE LAMINATES UNDER TENSION-COMPRESSION CYCLIC LOADS
NAG-1-232
82 January 1 - 84 January 1

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Principal Investigator: Dr. Wayne W. Stinchcomb
Department of Engineering Science and Mechanics
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-5316

Objective: To determine life-limiting fatigue damage mechanisms in graphite/epoxy laminates containing open holes and subjected to tension-compression fatigue loading.

DETERMINATION OF INTERLAMINAR FRACTURE TOUGHNESS OF UNIDIRECTIONAL COMPOSITES UNDER DYNAMIC CONDITIONS
NAG-1-347
83 May 16 - 84 May 15

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Principal Investigator: Dr. I. M. Daniel
Department of Mechanical Engineering
Illinois Institute of Technology
Chicago, Illinois 60616
(312) 567-3186

Objective: To determine whether interlaminar fracture toughness of brittle and "tough" composites is sensitive to load rate.

AN INVESTIGATION OF THE ACCURACY OF FINITE-DIFFERENCE METHODS IN THE SOLUTION OF LINEAR ELASTICITY PROBLEMS
NAG-1-316
83 January 1 - 83 December 31

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Principal Investigators: Dr. Nelson R. Bauld, Jr.
Dr. James G. Goree
Department of Mechanical Engineering
Clemson University
Clemson, South Carolina 29631
(803) 656-3470/3291

Objective: To identify potential problems in the application of finite-difference methods to elasticity problems involving stress singularities and discontinuities. To develop reliable techniques to cope with these problems.

ANALYSIS OF WOVEN FABRIC REINFORCED COMPOSITES
NAS1-17205
82 November 1 - 86 January 15

Project Engineer: H. Benson Dexter
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Spring House, Pennsylvania 19477
(215) 542-8400

Objective: To develop analytical methods to understand and predict the physical behavior of woven fabric reinforced composites, extend micromechanics methods to analysis of strength and toughness properties, evaluate potential of improved fabric designs, and develop guidelines for improved weaves. Included will be two-dimensional and three-dimensional woven fabrics with potential for improved fracture toughness and impact resistance.

EFFECTS OF SPECIMEN VARIABILITY AND MATERIAL DEFECTS ON THERMAL EXPANSION OF GRAPHITE/EPOXY COMPOSITES
NCC1-15
80 October 1 - 84 June 30

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2143 FTS 928-2143

Principal Investigator: Dr. M. W. Hyer
Department of Engineering Science and Mechanics
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-5905

Objective: To determine the effects of variability between specimens and material defects formed during fabrication on the thermal expansion of composite materials.

EFFECTS OF HIGH-ENERGY RADIATION ON THE MECHANICAL PROPERTIES OF GRAPHITE FIBER REINFORCED EPOXY RESINS
NSG-1562
79 October 1 - 83 December 31

Project Engineer: Dr. Edward R. Long, Jr.
Mail Stop 399
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3892 FTS 928-3892

Principal Investigators: Dr. Jasper D. Memory
Dr. Raymond E. Fornes
Departments of Physics and Textiles
North Carolina State University
Raleigh, North Carolina 27650
(919) 737-2503/3231

Objective: To investigate the effects of high-energy radiation on graphite fiber composites by study of composite curing effects, radiation exposure rates, mechanical fracture surfaces, and electron spin resonance properties.

ENVIRONMENTAL EXPOSURE EFFECT ON COMPOSITE MATERIALS FOR COMMECIAL AIRCRAFT
NAS1-15148
77 November 1 - 88 November 30

Project Engineer: Dr. Ronald K. Clark
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2143 FTS 928-2143

Principal Investigator: Martin Gibbons
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 655-4168

Objective: To provide technology in the area of environmental effects on graphite/epoxy composite materials, including long-term performance of advanced resin-matrix composite materials in ground and flight environments.

EFFECTS OF STRESS CONCENTRATIONS IN COMPOSITE STRUCTURES
NSG-1483
78 January 15 - 84 January 14

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Principal Investigators: Dr. Wolfgang G. Knauss
Dr. Charles D. Babcock
California Institute of Technology
Pasadena, California 91125
(213) 356-4524/4528

Objective: To study the effects of low-speed impact damage in composite structural components using high-speed motion pictures and to develop an analytical procedure for the propagation of the resulting impact damage.

ADVANCED COMPOSITE STRUCTURAL DESIGN TECHNOLOGY FOR COMMERCIAL TRANSPORT AIRCRAFT
NAS1-15949
79 September 24 - 84 September 23

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Principal Investigator: John N. Dickson
Lockheed-Georgia Company
86 South Cobb Drive
Marietta, Georgia 30063
(404) 424-3085

Objective: To design, analyze, fabricate, and test generic advanced-composite structural components for transport aircraft applications in order to develop verified design technology.

STRUCTURAL OPTIMIZATION FOR IMPROVED DAMAGE TOLERANCE
NAG-1-168
81 September 1 - 84 October 15

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Principal Investigator: Dr. Raphael T. Haftka
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-4860

Objective: To develop a structural optimization procedure for composite wing boxes that includes the influence of damage-tolerance considerations in the design process.

EVALUATION OF THE DURABILITY AND DAMAGE TOLERANCE OF COMPOSITE STRUCTURES SUITABLE FOR COMMERICAL TRANSPORT AIRCRAFT
NAS1-15107
77 October 1 - 84 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3524 FTS 928-3524

Principal Investigator: Robert D. Wilson
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 655-4127

Objective: To design, fabricate, and test generic composite structural components for commercial aircraft applications that are durable and damage tolerant.

COMPRESSION FAILURE MECHANISMS OF COMPOSITE STRUCTURES
NAG-1-295
82 September 1 - 84 August 31

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3524 FTS 928-3524

Principal Investigator: Dr. H. Thomas Hahn
Washington University
Campus Box 1087
St. Louis, Missouri 63130
(314) 889-6052

Objective: To establish the effects of material properties on microbuckling and the shear crippling failure mode in order to design stronger, more damage tolerant composite structures.

DEFORMATION MEASUREMENTS OF COMPOSITE MULTI-SPAN BEAM SHEAR SPECIMENS BY MOIRE INTERFEROMETRY
NAG-1-359
83 May 1 - 83 December 31

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3524 FTS 928-3524

Principal Investigator: Dr. Daniel Post
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-6651

Objective: To accurately measure the transverse deformations and strains of a short multiple-span composite beam for comparison with theoretical predictions.

FABRICATION AND ANALYSIS OF STITCHED GRAPHITE/EPOXY LAMINATES
NAG-1-381
83 July 1 - 83 December 31

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3524 FTS 928-3524

Principal Investigator: Dr. C. T. Sun
School of Aeronautics and Astronautics
Purdue University
West Lafayette, Indiana 47907
(317) 494-5130

Objective: To assess the merit, by analysis and experiment, of angular stitching on the transverse shear stiffness and strength of composite laminates.

NASA LEWIS RESEARCH CENTER

INHOUSE

SIMPLIFIED COMPOSITE MICROMECHANICS EQUATIONS
80 October 1 - 83 September 30

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6138

Objective: Develop composite micromechanics equations with and without interface for predicting hygrothermomechanical properties and validate predicted results with finite element analysis.

FINITE ELEMENT SUBSTRUCTURING FOR COMPOSITE MECHANICS
82 September 15 - 84 December 30

Project Engineers: John J. Caruso/Pappu L. N. Murthy
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5366

Objective: Develop super element finite element models for describing composite micromechanics behavior and stress concentrations in angleplied laminates.

CRACKED COMPOSITE CHARACTERIZATION
82 July 7 - 85 September 30

Project Engineers: Thomas B. Irvine/Carol A. Ginty
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5367

Objective: Conduct experimental/theoretical investigations using RUSCAN/CODSTRAN (Real-Time Ultrasonic C-Scanning/Composite Durability Structural Analysis) to characterize progressive fracture and attendant failure modes in fiber composites with and without defects and subjected to hygrothermomechanical environments.

CODSTRAN-CONTINUING DEVELOPMENT
82 July 7 - 85 December 31

Project Engineer: Thomas B. Irvine
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5367

Objective: Continue development/documentation of CODSTRAN (Composite Durability Structural Analysis) with respect to participating fracture modes, combined stress failure criteria, complex loading conditions and corroboration with experimental data.

FAILURE MODES AND FRACTURE SURFACE CHARACTERISTICS
80 September 1 - 84 December 15

Project Engineer: Carol A. Ginty
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Objective: Identify/characterize failure modes and attendant fracture surface characteristics in angleplied laminates subjected to uniaxial and combined loads.

LIFE/DURABILITY IN HYGROTHERMOMECHANICAL ENVIRONMENTS
81 June 1 - 84 December 31

Project Engineers: Carol A. Ginty/Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Objective: Continue application and experimental corroboration of Lewis-developed hygrothermomechanical theory to different fiber composites and under various adverse loading conditions.

ICAN-INTEGRATED COMPOSITE ANALYSIS COMPUTER CODE
82 October 15 - 84 September 14

Project Engineers: Pappu L. N. Murthy/Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Objective: Develop/document a general purpose, integrated computer program (code) for fiber composite structural/stress analysis and for composite mechanics.

HYGROTHERMOMECHANOCHRONIC THEORY
83 March 1 - 86 April 30

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Objective: Develop a unified hygro-thermo-mechano-chronic (time) theory to predict the hygrothermomechanochronic behavior of fiber composites including damping, temperature rise due to damping and attendant degradation effects, and corroborate with experimental data.

N. L. COBSTRAN
82 January 4 - 85 September 30

Project Engineer: Dale A. Hopkins
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5366

Objective: Extend COBSTRAN (Composite Blade Structural Analysis) to nonlinear thermoviscoplastic structural analysis for high temperature fiber composite turbine blades.

ANALYSIS OF ADVANCED TURBOPROPS
81 January 15 - 84 December 31

Project Engineer: Robert A. Aiello
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6272

Objective: Use COBSTRAN to predict the structural behavior of advanced swept turboprops made with a composite-shell and metal-spar and to conduct parametric studies for the influence of composite system and laminate configuration on structural behavior.

HIGH VELOCITY IMPACT RESISTANCE OF ALUMINUM MATRIX COMPOSITES
80 October 15 - 83 November 30

Project Engineer: David L. McDanels
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6956

Objective: To determine the high velocity impact damage threshold of several aluminum component materials and to relate the relative energy absorption mechanisms to material variables.

FAILURE MODES OF TUNGSTEN FIBER REINFORCED SUPERALLOYS
81 October 1 - 85 September 30

Project Engineer: Donald W. Petrasek
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6284

Objective: To evaluate the failure of TFRS specimens subjected to combined cyclic stress and temperature conditions as well as steady state stress and temperature conditions to develop failure models to predict performance.

IMPROVED TOUGHNESS HIGH TEMPERATURE RESINS
83 October 1 - 84 October 30

Project Engineer: Kenneth J. Bowles
MS 49-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6967

Objective: To achieve a fundamental understanding of the factors which control the toughness characteristics of high temperature polymer matrix composites and to evolve criteria for predicting composite performance.

ULTRASONIC ASSESSMENT OF SHUTTLE FILAMENT - WOUND CASE (FWC) MATERIAL
83 April 1 - 84 March 31

Project Engineer: Alex Vary
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6357

Objective: Study applications of backscatter, pulse-echo, and acousto-ultrasonic approaches to assessment of initial and post-use state of FWC material with emphasis on assessment of degradation/reusability.

CERAMIC MATRIX COMPOSITES
81 September 30 - 84 September 30

Project Engineer: Dr. H. H. Grimes
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6601

Objective: To develop and evaluate processing methods for the preparation of ceramic matrix composites reinforced by continuous ceramic fibers to provide new, advanced materials for aerospace applications.

ADVANCED COMPOSITE MICROMECHANICS
81 September 30 - 84 September 30

Project Engineer: Dr. H. H. Grimes
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6601

Objective: To determine the principal macro- and microstructural factors which control deformation, strength and toughness of advanced composite materials.

NASA LEWIS RESEARCH CENTER

CONTRACTS/GRANTS

DYNAMIC DELAMINATION
NAG 3-211
81 December 15 - 84 December 14

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Principal Investigator: C. T. Sun
School of Aeronautics and Astronautics
Purdue University
West Lafayette, Indiana 47907
(317) 494-5125

Objective: Develop analytical/experimental methods to describe and characterize dynamic interlaminar delamination propagation in fiber composites.

TEST METHODS AND CHARACTERIZATION OF HIGH TEMPERATURE COMPOSITES
NAG 3-377
82 December 10 - 86 December 9

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Principal Investigator: John F. Mandell
Department of Materials Science and Engineering
Massachusetts Institute of Technology
Cambridge, Massachusetts 02139
(617) 253-7181

Objective: Develop test methods and characterize the thermomechanical behavior of high temperature fiber composites.

ADVANCED COMPOSITE COMBUSTOR STRUCTURAL CONCEPTS
NAS 3-23284
81 April 1 - 83 December 31

Project Engineer: Robert L. Thompson
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5366

Principal Investigator: Robert P. Lohmann
Pratt and Whitney Aircraft
400 Main Street
East Hartford, Connecticut 06108
(203) 565-7778

Objective: Conduct preliminary design and evaluation study of an advanced combustor using high temperature composite materials.

EFFECTS OF ENVIRONMENT AND DEFECTS ON HIGH STRAIN RATE PROPERTIES OF COMPOSITES
NAG 3-423
83 May 15 - 86 May 14

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Principal Investigator: Isaac M. Daniel
Mechanical Engineering Department
Illinois Institute of Technology
Chicago, Illinois 60616
(312) 567-3186

Objective: Develop experimental procedures to study the influence of environment (moisture and temperature) and defects of the high-strain-rate properties of fiber composites.

STRUCTURAL DESIGN STUDY OF LOW SPEED PROPELLERS
NAS 3-23924
83 April 22 - 84 October 21

Project Engineer: Robert A. Aiello
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6272

Principal Investigator: Bennett M. Brooks
Hamilton Standard
Windsor Locks, Connecticut 06906
(203) 623-1621, ext. 5611

Objective: Identify the most promising propeller configurations incorporating advanced concepts and materials and provide optimized designs.

STAEBL
NAS 3-22525
80 September 30 - 85 December 31

Project Engineer: Murray S. Hirschbein
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6272

Principal Investigator: Kenneth W. Brown
Pratt and Whitney Aircraft
400 Main Street
East Hartford, Connecticut 06108
(203) 565-7053

Objective: Develop a formalized optimum design procedure for engine blades made using advanced structural concepts and materials and meet all the aerothermomechanical design requirements in aircraft engine environments.

ANALYSIS OF HIGH VELOCITY BALLISTIC IMPACT RESPONSE OF BORON/ALUMINUM FAN BLADES
81 October 15 - 83 December 31

Project Engineer: D. L. McDanels
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6956

Principal Investigator: R. Ravenhall
General Electric Co.
Aircraft Engine Group
Cincinnati, Ohio 45215
(513) 243-2000

Objective: Correlate high velocity impact test data from B/Al specimens using computer model analysis to predict performance as a fan blade.

ULTRASONIC STRESS WAVES CHARACTERIZATION OF COMPOSITE MATERIALS
82 October 1 - 83 September 30

Project Engineer: Alex Vary
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6357

Principal Investigators: E. G. Henneke, II
J. C. Duke,, Jr.
W. W. Stinchcomb
Engineering Science & Mechanics Department
Virginia Polytechnic Institute
Blacksburg, Virginia 24061
(703) 961-5316

Objective: Establish relations between the acousto-ultrasonic stress wave factor (SWF) and mechanical response of composite laminates with emphasis on degradations due to cyclic fatigue.

INVESTIGATION OF INTERFACIAL PHASE FORMATION IN FIBER REINFORCED CERAMIC MATRIX COMPOSITE MATERIALS
83 March 1 - 84 March 1

Project Engineer: Dr. D. R. Behrendt
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6602

Principal Investigator: Dr. F. E. Wawner
University of Virginia
Charlottesville, Virginia 22901

Objective: To fabricate and fully characterize low density, high performance fiber reinforced ceramic matrix composite materials with respect to structural and microstructural responses to thermal treatment.

NAVAL AIR SYSTEMS COMMAND
WASHINGTON, D.C. 20361

INHOUSE

FATIGUE OF COMPOSITES UNDER COMPLEX LOADS
79 October - 83 September

Project Engineer: Dr. P. W. Mast
Naval Research Laboratory
Washington, D.C. 20375
(202) 767-2165 Autovon 297-2165

Objective: Develop a capability for predicting the structural response and initiation of failure in composite laminates and bonded joints under complex cyclic loading.

REPAIR OF COMPOSITE LAMINATES
82 March - 84 September

Project Engineer: Dr. J. Augl
Naval Surface Weapons Center
White Oak, Silver Spring, MD 20910
(204) 394-2262 Autovon 290-2261

Objective: To develop models for diffusion of moisture in composite laminates during thermal cycling associated with bonded repairs.

CONTRACTS

DELAMINATION FAILURE CRITERIA FOR COMPOSITE STRUCTURES
82 September - 84 March

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-7447 Autovon 222-7447

Principal Investigator: Dr. R. Wilkins
General Dynamics
Fort Worth, TX 76101
(817) 732-4811 Ext. 4631

Objective: Conduct experimental studies to develop a delamination failure criteria and analysis methods to predict debonding in composite structures.

Enclosure (1)

DELAMINATION IN COMPOSITE STEPPED LAP JOINTS
80 August - 83 June

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-7447 Autovon 222-7447

Principal Investigator: Dr. M. M. Ratwani
Northrop Corporation
Hawthrone, CA 90250
(213) 970-5285

Objective: Conduct analytical and experimental studies of delamination in a laminated composite metallic stepped lap joint configuration.

FATIGUE LIFE AND RESIDUAL STRENGTH OF COMPOSITE STRUCTURES
83 September - 85 September

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-7447 Autovon 222-7447

Principal Investigators: Dr. J. Yang and
Dr. D. Jones
The George Washington University
Washington, D.C. 20052
(202) 676-6929

Objective: Develop statistical models to describe fatigue life and residual strength of composite structures including bolted and bonded composite joints.

DELAMINATION OF COMPOSITE SKIN STIFFENED STRUCTURES
82 September - 83 December

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-7447 Autovon 222-7447

Principal Investigator: Mr. Ronald Knight
LTV Advanced Technology Center
Dallas, TX 75260 and
Dr. Edmund Rybicki
Univ. of Tulsa
Tulsa, OK 74104

Objective: To develop fracture mechanics models to describe delamination between hat-stiffeners and skins for a bonded composite fuselage structure.

NAVAL AIR DEVELOPMENT CENTER
AIRCRAFT AND CREW SYSTEMS TECHNOLOGY DIRECTORATE
WARMINSTER, PA 18974

INHOUSE

COMPOSITE IMPACT RESISTANCE
74 March - 84 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

NRC Guest Investigator: Dr. P. V. McLaughlin
Villanova University
Villanova, PA 19085

Objective: Ascertain the impact response of generic composite structural elements and identify the physical mechanisms associated with impact damage and the critical parameters governing impact response.

HYBRID COMPOSITE FRACTURE CHARACTERIZATION
80 September - 84 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Objective: Characterize the strength, mechanical properties, and failure characteristics of woven and intimately mixed hybrid composite laminates.

ANALYTICAL MODELING OF COMPOSITE FAILURE MODES
83 October - 84 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(214) 441-2867 Autovon 441-2867

NRC Guest Investigator: Dr. A.S.D. Wang
Drexel University
Philadelphia, PA 19104

Objective: Develop a mixed-mode crack growth criterion for delamination growth and develop the load-damage-life relationship concept for describing the failure process in laminates.

CONTRACTS

DESIGN OF HIGHLY LOADED COMPOSITE JOINTS
AND ATTACHMENTS FOR TAIL STRUCTURES
N62269-82-C-0239
82 February - 84 July

Project Engineer: Ramon Garcia
Naval Air Development Center
ACSTD/60432
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: S. W. Averill
Northrop Corporation
Aircraft Group
Hawthorne, CA 90250
(213) 970-3442

Objective: To develop composite designs which will permit the use of metal to composite bolted root attachments in aircraft tail structures as an alternative to high-load transfer adhesive bonded titanium step joints. To improve damage tolerance, survivability and repairability over current composite designs. Structural efficiency, manufacturing feasibility and quality assurance requirements will be determined.

DESIGN OF HIGHLY LOADED COMPOSITE JOINTS
AND ATTACHMENTS FOR WING STRUCTURES
N62269-82-C-0238
82 February - 85 April

Project Engineer: Ramon Garcia
Naval Air Development Center
ACSTD/60432
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: M. J. Ogonowski
McDonnell Aircraft Co.
P. O. Box 516
St. Louis, MO 63166
(314) 233-8630

Objective: To develop composite designs which will permit the use of metal to composite bolted root attachments in aircraft wing structures as an alternative to high-load transfer adhesive bonded titanium step joints. Strain concentration around fastener holes, fatigue and environmental affects, damage tolerance and repairability for each concept will be determined.

QUASI 3-DIMENSIONAL FINITE ELEMENT ANALYSIS OF DELAMINATION GROWTH IN COMPOSITES
N62269-82-C-0250
82 April - 83 October

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. A. S. D. Wang
Drexel University
Philadelphia, PA 19104
(215) 895-2297

Objective: Linear elastic fracture mechanics will be generalized to the more complicated, two-dimensional delamination process by providing a convenient computational scheme to accurately determine the three-dimensional stress field surrounding a delamination crack and a general delamination growth criterion developed.

POLYMER MATRIX FATIGUE PROPERTIES
N62269-80-C-0278
80 September - 83 June

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. D. F. Adams
Univ. of Wyoming
Laramie, WY 82071
(307) 766-2371

Objective: Characterize and compare the fatigue properties of 3501 and X4001 neat resin materials at room temperature and 88°C.

SUPPRESSION OF DELAMINATION IN COMPOSITES BY THICKNESS DIRECTION REINFORCEMENT
N62269-82-C-0248
82 June - 83 December

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. C. T. Sun
Purdue University
West Lafayette, IN 47907
(317) 494-5130

Objective: Improve the damage tolerance and durability of laminated composite structure by providing thickness-direction reinforcement to constrain the growth of delamination damage.

DEVELOPMENT OF HIGH STRAIN COMPOSITE WING
N62269-81-C-0727
81 September - 84 March

Project Engineer: Mark Libeskind
Naval Air Development Center
ACSTD/60433
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: J. Bruno
Grumman Aerospace Corporation
Bethpage, NY 11714
(516) 575-6295

Objective: Design and evaluate an advanced composite wing which operates at significantly higher strain levels than current composite wings resulting in significant weight savings. Emphasis will be placed upon damage tolerance, survivability, durability and repairability.

ARMY MATERIALS AND MECHANICS RESEARCH CENTER

INHOUSE

JOINT DESIGN METHODOLOGY FOR COMPOSITE STRUCTURES
75 October 1 - 87 September 30

Project Engineer: D. W. Oplinger
Army Materials and Mechanics Research Center
ATTN: DRXMR-SME
Watertown, MA 02172
(617) 923-5166 Autovon 955-5166

Objective: To investigate the structural performance of mechanically fastened and bonded joints using advanced analytical and experimental (Moire, laser speckle, etc.) techniques and to develop new joint design procedures for composite structures based on the results of such efforts.

VULNERABILITY OF JOINTS IN COMPOSITE STRUCTURES

78 October 1 - 86 September 30

Project Engineer: D. W. Oplinger
Army Materials and Mechanics Research Center
ATTN: DRXMR-SME
Watertown, MA 02172
(617) 923-5166 Autovon 955-5166

Objective: Examine structural degradation effects of 23mm HEI and similar threats causing combined blast/fragment damage in composite structures. Develop joint and structural concepts for hardening composite structural joints against such threats.

EVALUATION OF FRICTION JOINT CONCEPTS IN COMPOSITE STRUCTURES
79 October 1 - 85 September 30

Project Engineer: Dr. J. Slepetz
Army Materials and Mechanics Research Center
ATTN: DRXMR-SME
Watertown, MA 02172
(617) 923-5746 Autovon 955-5746

Objective: Design and experimentally demonstrate joint concepts for composite structures using clamping friction as a means of load transfer to avoid stress concentrations normally associated with bolted joints. Investigate problems associated with bolt tension relaxation and other effects related to reliability of friction joint concepts.

EFFECTS OF DAMAGE ZONES ON FAILURE BEHAVIOR OF NOTCHED COMPOSITE LAMINATES
80 October 1 - 86 September 30

Project Engineer: Dr. J. Slepetz
Army Materials and Mechanics Research Center
ATTN: DRXMR-SME
Watertown, MA 02172
(617) 923-5746 Autovon 955-5746

Objective: Investigate microstructure of damage zones produced by structural loading of notched composites. Develop simplified approaches for modeling damage zone behavior in finite element analysis and use the results of such studies to provide simplified failure criteria for use by designers of composite structures.

DEVELOPMENT OF STATISTICAL RELIABILITY METHODOLOGY FOR COMPOSITE MATERIALS
80 October 1 - 87 September 30

Project Engineer: D. M. Neal
Army Materials and Mechanics Research Center
ATTN: DRXMR-SME
Watertown, MA 02172
(617) 923-5165 Autovon 955-5165

Objective: Investigate statistical characteristics of failure data for composite materials under tension, compression, shear and fatigue loading and develop improved approaches for obtaining allowables and other reliability-related parameters of composite materials.

MECHANICS OF LASER-STRUCTURE INTERACTION
81 October 1 - 86 September 30

Project Engineer: J. Adachi
Army Materials and Mechanics Research Center
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Objective: Identify material properties data needs for determining vulnerability of composite structures to laser exposure. Assess vulnerability assessment codes. Assess relative hardness of structural concepts.

EVALUATION OF IN-SERVICE DURABILITY OF COMPOSITE STRUCTURES
75 October 1 - 86 September 30

Project Engineer: Dr. M. Roylance
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Objective: Evaluation of mechanical property degradation of glass and Kevlar composites under enviornmental exposure, for static and fatigue loading. Application of advanced evaluation techniques such as automated dynamic structural response (modal analysis) for assessing degradation effects.

INVESTIGATION OF PREDICTIVE CHARACTERIZATION TECHNIQUES FOR COMPOSITES
81 October 1 - 88 September 30

Project Engineer: R. Shufford
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Objective: Evaluation of advanced characterization techniques for composite materials such as the application of vibrothermography, acoustic emission and creep for monitoring curing steps, especially in graphite reinforced materials. Investigation of tensile characteristics.

DEVELOPMENT OF ADVANCED NUMERICAL ANALYSIS TECHNIQUES FOR FAILURE PREDICTION IN COMPOSITE MATERIALS
82 August 1 - 86 September 30

Project Engineer: Dr. R. Barsoum
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Objective: Develop improved finite element methodology for analyzing edge effects and incipient failure in composite laminates.

DATA FITTING METHODOLOGY FOR MECHANICAL TESTING OF COMPOSITES
80 October 1 - 86 September 1

Project Engineer: R. Papirno
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Objective: Invensive review of stress-strain data from mechanical testing of composite materials and development of efficient methods of representation of such data for purposes of computerized data bank generation

IMPROVED TENSION TEST SPECIMENS FOR COMPOSITE MATERIALS
77 October 1 - 86 September 30

Project Engineer: D. W. Oplinger
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Objective: Investigate the performance characteristics of available tension test specimens using combined stress analysis and experimental approaches. Develop streamline tension specimen shapes and evaluate their performance collaborative efforts with ASTM D30.

IMPROVED SHEAR TESTING METHODS FOR COMPOSITE MATERIALS
76 October 1 - 88 September 30

Project Engineer: Dr. J. Slepetz
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Objective: Evaluate available methods for in-plane shear testing of composite materials. Development of improved methods using Iosepescu-type specimen geometry in conjunction with antisymmetric four point loading fixture.

CONTRACTS

ANALYSIS OF ADHESIVELY BONDED JOINTS IN COMPOSITE AND METALLIC STRUCTURES
DAAG46-82-K-0025
82 March 1 - 85 February 28

Project Engineer: D. W. Oplinger
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Principal Investigator: Professor J. Vinson
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Objective: Develop stress analysis methods for bonded joints having improved efficiency over conventional finite element methods. Experimental investigation of effects of through-thickness variations of bond layer properties using acoustic shear wave measurements in combination with progressive sectioning of bond layers. Investigation of viscoelastic and fracture behavior of adhesive layers.

MICROGRAPHIC EXAMINATION OF STRUCTURAL DAMAGE IN COMPOSITE MATERIALS
DAAG-46-81-C-0010
81 October 1 - 85 September 30

Project Engineer: Dr. J. Slepetz
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Principal Investigator: Professor J. Mandel
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Objective: Experimental evaluation of the structure of damage zones in laminates containing crack-like notches as well as in pin loaded coupons representative of bolted joints. Micrographic characterization of the structure of the damage and development of methods for relating such damage to simplified damage zone representations in failure-rule development efforts.

VISCOELASTIC BEHAVIOR CHARACTERIZATION OF KEVLAR EPOXY MATERIALS
DAAG46-83-C-0032
83 September 1 - 85 August 31

Project Engineer: Dr. M. Roylance
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Principal Investigator: T. L. Ho
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Objective: Characterize viscoelastic response of kevlar epoxy materials under various environmental conditioning situations.

☆U.S. GPO: 1983-659-006/311

MECHANICS OF COMPOSITES REVIEW
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24-26 OCTOBER 1983

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